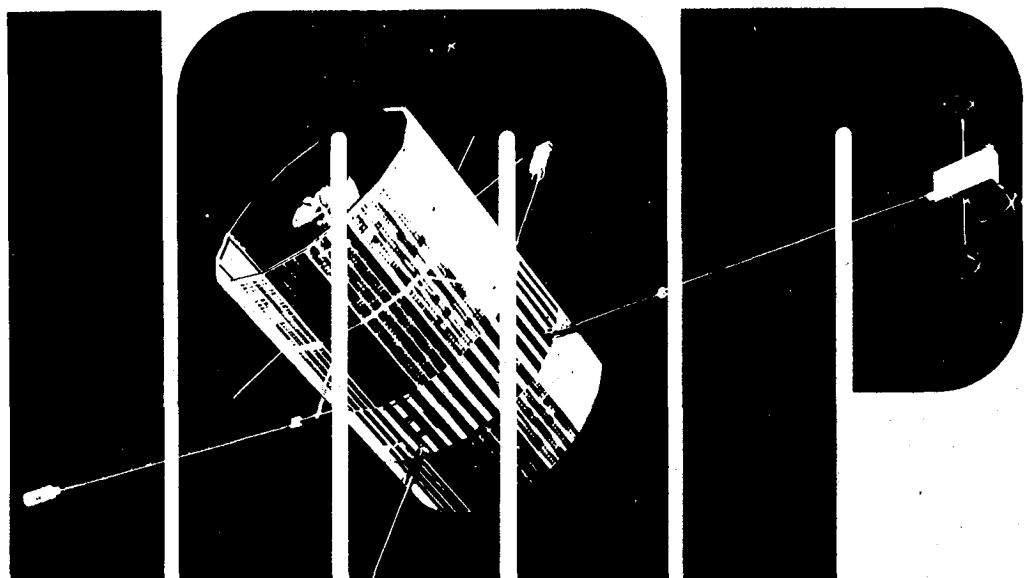


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Interplanetary Monitoring Platform



Engineering History and Achievements

NASA

National Aeronautics and
Space Administration

**Goddard Space Flight Center
Greenbelt, Maryland**

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Interplanetary Monitoring Platform

Engineering History and Achievements

May 1980



National Aeronautics and
Space Administration

**Goddard Space Flight Center
Greenbelt, Maryland**

FOREWORD

In November 1975, the last of ten Interplanetary Monitoring Platform (IMP) Satellites ended a series of pioneering scientific observations dedicated to obtaining data during a complete solar cycle (11 yrs.) to enhance man's knowledge of the radiation environment of cislunar space. The scientific achievements and the technological accomplishment of the IMP program greatly expanded our understanding of the Sun-Earth relationship and paved the way both scientifically and technologically for the International Sun Earth Explorer (ISEE) program.

During the IMP program, major scientific discoveries and technological advancements were accomplished. By careful planning the Project managed to take advantage of the latest technological advances in the Nation to support the pioneering efforts of the scientists in the universities, government, and industry.

The success of the IMP program is a tribute to the talent, hard work, and long hours dedicated by 300-400 of the Nation's best scientists and aerospace engineers in the universities, government, and industry. Only by this team effort over an 11-year period could so much be accomplished by so few for so little. There are many unsung heroes who will be long remembered.

February 1980

Paul Butler
Interplanetary Monitoring
Platform Project Manager
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Chapter 1

The IMP Program-An Overview

INTRODUCTION

In the late 1950s and early 1960s, the radiation hazards that might be encountered in manned space travel were a major concern to the National Aeronautics and Space Administration (NASA) since the safety of our astronauts was of paramount importance to the success of the Nation's space program. This overriding concern brought about an extensive NASA effort directed toward achieving a better understanding of space and its phenomena and the development of new technology to improve our spacecraft and scientific missions.

In the Spring of 1961, NASA's Goddard Space Flight Center (GSFC) in Greenbelt, Maryland recommended to NASA Headquarters that a series of monitoring probes or satellites be developed which could continuously measure the radiation environment in the immediate vicinity of the Earth and in interplanetary space in preparation for the Apollo program which would eventually put man on the Moon.

By making continuous measurements in space, NASA scientists and engineers could define the conditions which the astronauts would encounter, and at the same time, provide scientists with new knowledge about phenomena such as magnetic fields, plasma, and energetic particles in space.

The project was designated the Interplanetary Monitoring Probe (IMP). It was initiated by NASA in the Fall of 1961. A year later, the name was changed to the Interplanetary Monitoring Platform. A series of ten IMP satellites were developed and launched over the next ten-year period beginning with IMP A in November 1963 (Figure 1).

The Interplanetary Monitoring Platform (IMP) Satellites Program may be considered one of the stepping stones to manned space travel and to the Nation's greatest space achievement—the landing of U.S. astronauts on the Moon. At no time in its initial planning process did the IMP satellites attain the possibility of standing alone in its potentiality of scientific discovery. Like other NASA efforts which were dedicated

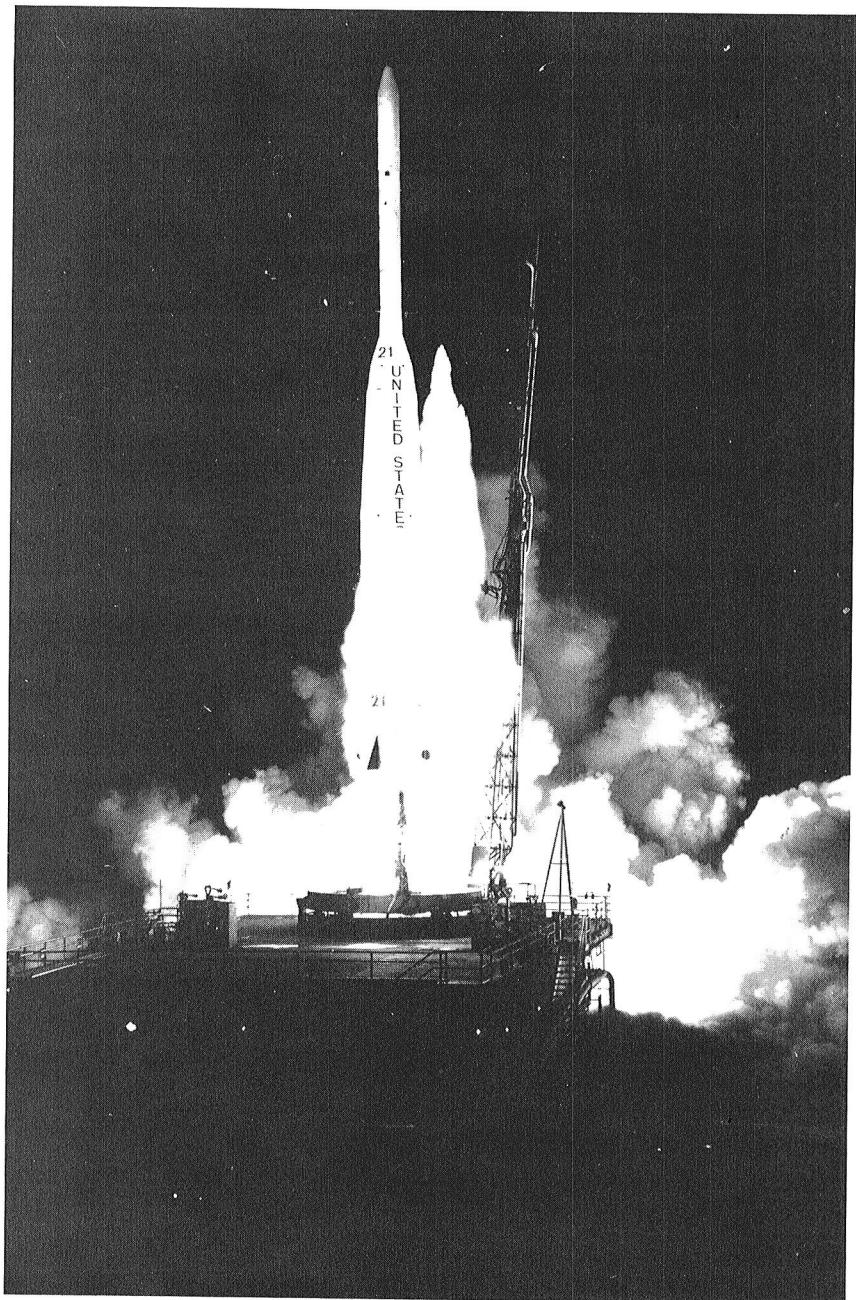


Figure 1. The IMP A Spacecraft, First in a Series of Ten Spacecraft, was Launched by the Delta 21 Vehicle on November 26, 1963.

to obtaining data on outer space phenomena and potential problems for astronauts, the IMP satellite series made a number of important discoveries and contributions even as late as 1978. For example, the IMP spacecraft provided direct measurement of the radiation in space during the Apollo flights and produced real-time data on solar flare activity during the mission. In other detailed measurements of the near interplanetary environment, the IMP spacecraft were highly successful in furthering our understanding of the Earth's radiation environment and solar-terrestrial physics. Nearly continuous interplanetary measurements of magnetic fields, plasmas, and energetic solar and galactic particles were made over the years. In addition, IMP measurements taken in distant magnetospheric regions, the bow shock, the magnetosheath, the magnetopause, and the magnetotail were crucial to building an understanding of those regions and rewriting of previous textbook explanations of the phenomena in outer space.

The program also contributed a number of important firsts in the development of spacecraft technology during its 10-year life span. IMP A was the first satellite to use the modular concept of packaging for experiment instruments and their associated electronic subsystems. Another important technology achievement was in the first use of integrated circuits in a spacecraft and in the use of an on-board computer for data handling.

Other major contributions in spacecraft technology included the development of spin and despin attitude control systems which allowed the reorientation of the spacecraft to be accomplished from commands on the ground.

Accomplishments also include the evolutionary development of spacecraft power systems ranging from 38 watts of power in the first spacecraft in the series to over 130 watts of available power for the last group of spacecraft. IMP program accomplishments also included the development of a procedure for the biological decontamination of spacecraft systems and components, and the development of standard methods of designing, fabricating and testing magnetically clean spacecraft. These methods were used in later space programs, such as the Pioneers 10 and 11 probes to Jupiter.

These are just some of the major IMP program achievements which characterize the contribution of the joint Government, university, and industry team and the spacecraft missions which are detailed in this book. Detailed accounts of the results from the various scientific experiments flown on the series of IMP satellites are scheduled to be published in the near future.

PROGRAM OBJECTIVES

From the outset of the program, NASA planners desired to bring together NASA expertise, the university and scientific community, and industry in a major joint or cooperative project. In this cooperative arrangement, NASA could have available to it the best possible scientists and engineers working toward achieving the overall objectives of the program. Headed by members of the GSFC Engineering Directorate, this team of experts established a number of important objectives which could lead to an advancement in spacecraft technology and scientific knowledge and instrumentation.

The scientific objective of the IMP missions was to significantly advance man's knowledge of interplanetary space and its phenomena by making direct measurements from orbits around the Earth and the Moon. Particular emphasis was placed on the following major objectives:

- Study the radiation environment in the Earth's magnetosphere and in cislunar space, and monitor the radiation in space in support of the Apollo program;
- Study the quiescent properties of the interplanetary magnetic fields and their relationship with the particle fluxes emanating from the Sun;
- Develop a capability for predicting periods of solar-flare activities;
- Study solar-terrestrial relationships; and
- Develop spin-stabilized spacecraft which could be used as platforms containing experiments for obtaining required scientific data.

A number of project-oriented objectives also were established early in the program to be used as guidelines during the development process. These included goals ranging from implementation of new engineering management concepts which could be used by GSFC project managers as vehicles for running an improved project to mission-related goals which could enhance the mission operation, performance or data collection process.

These development objectives included the following types of activity:

- To develop the first IMP spacecraft using GSFC's own management and engineering staff and expertise;
- To procure the IMP hardware components from industry contractors utilizing the latest technology and techniques available;
- To make the IMP a low-cost, short lead-time spacecraft which could become a part of the Explorer satellite series supporting NASA science missions;
- To establish a quick launch-turnaround capability for the satellites

within a group so that experiments on subsequent flights could be optimized to provide data as efficiently and rapidly as possible;

- To select the best possible scientific experiments and payloads for each group of flights from throughout the Government, university, and industrial scientific community;
- To develop spacecraft environmental acceptance tests and verify adequacy by orbital flight performances.
- To employ the latest state-of-the-art instrumentation and experiment subsystems for each IMP satellite and to potentially upgrade or refine the experiment on subsequent flights based on data from prior flights;
- To schedule spacecraft launchings so that one satellite would be operational continuously during an 11-year solar cycle; and
- To ensure that the IMP design keeps pace with the development of more powerful rockets so that larger spacecraft could be launched which could carry additional or larger experiments. This would necessitate a continuous research and development approach to each new series of IMP spacecraft and instrument payload.

THE IMP MISSIONS

The IMP spacecraft were grouped by their characteristics and development stages. The IMPs A, B, and C formed one group, the “Anchored” IMPs the second, IMPs F and G another, IMP I the fourth, and the last two, IMPs H and J were the fifth group.

Because the entire IMP series of spacecraft was a subset of the Explorer series and because two of the IMPs were to have been “Anchored” to the same orbit as the Moon, the names of the IMP spacecraft may be confusing. For clarification, a list of names and launch dates is presented in Table 1.

Before their launch, the IMP spacecraft were known by their letter codes; after launch, an Explorer number and an IMP (or AIM) number was assigned. In many cases, a spacecraft is referred to by its prelaunch designation even after launch. Because the prelaunch designations constitute the most continuous sequence of names in the IMP series, reference to individual spacecraft by their prelaunch lettercode designations is made throughout this report. IMPs A, B, and C were the first spacecraft in the series to be developed (Figure 2). The spacecrafts were placed in eccentric orbits to map the Earth’s radiation environment at altitudes from 200 to 260,000 kilometers (km) during the first phase of an 11-year solar cycle. Designed primarily to carry out pioneering explo-

Table 1
IMP Spacecraft Names and Launch Dates

IMP	Explorer	IMP	AIMP	AIMP	Launch Date
A	18	1			11/27/63
B	21	2			10/04/64
C	28	3			05/29/65
D	33		D	1*	07/01/66
E	35		E	2**	07/19/67
F	34	4			05/24/67
G	41	5			06/21/69
I	43	6			03/13/71
H	47	7			09/22/72
J	50	8			10/25/73

*Lunar orbit intended but not achieved.

**Lunar orbit achieved.

ration in the near-Earth region and the outer portion of the Earth's magnetosphere, this first series of spacecraft provided valuable data which was used to study the interactions of the Earth-Sun system.

The second group, designated "Anchored" IMPs (AIMPs) D and E (Figure 3), were designed primarily for an extended study and mapping of interplanetary magnetic fields and for measuring solar plasma, energetic particles and micrometeorite fluxes in the vicinity of the Moon. They also were designed to measure the lunar gravitational field. Since lunar orbits were planned, the term "anchored" was used to distinguish these satellites from the other IMP satellites in the series which were to be in Earth orbits. The unique feature of the lunar orbit placed the satellite alternately in the Sun side and dark side of the Earth each month, at lunar altitudes. The highly eccentric Earth orbits take one year to "sweep out" the same region.

The third group of spacecraft in the series consisted of IMPs F and G (Figure 4). The primary objective of these satellites was to continue the study of the Earth's radiation environment with second-generation or improved experiments. These satellites were operated during a period of maximum solar activity, which was of particular importance to the scientific community. Data obtained during this period of maximum activity made a major contribution to man's knowledge of the radiation hazards in space and provided new insights into the phenomena of solar flares.

THE IMP PROGRAM—AN OVERVIEW

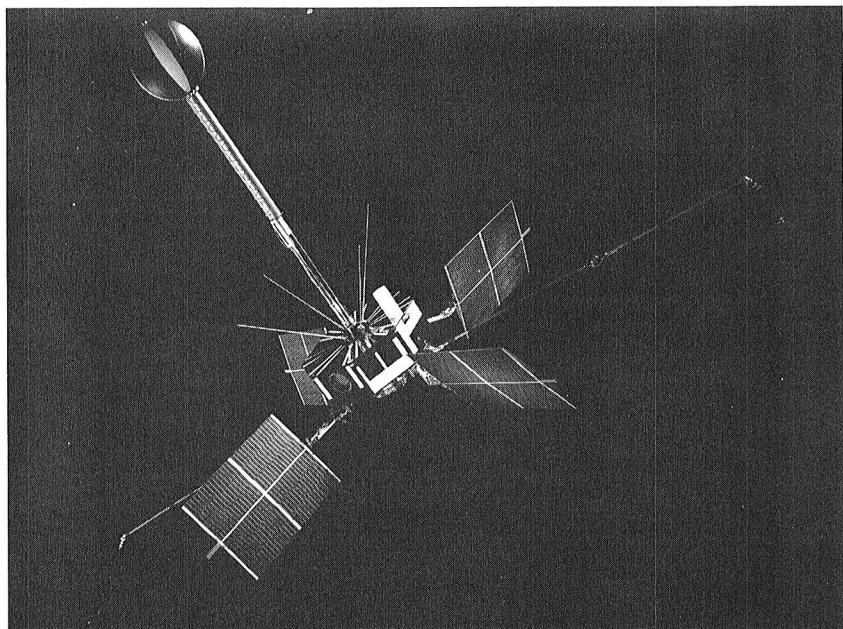


Figure 2. IMP A Spacecraft in Orbital Configuration.

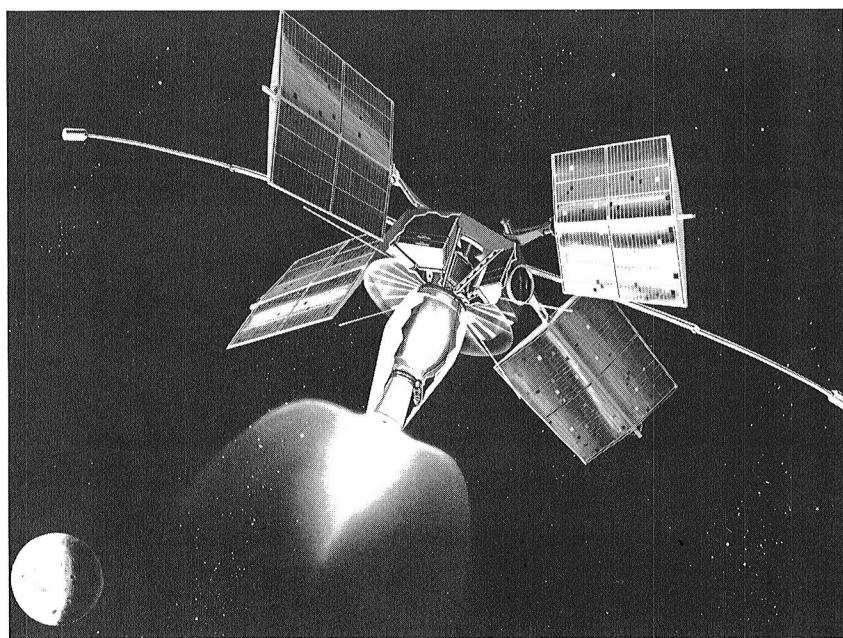


Figure 3. Artist Conception of AIMPS D and E Spacecraft.

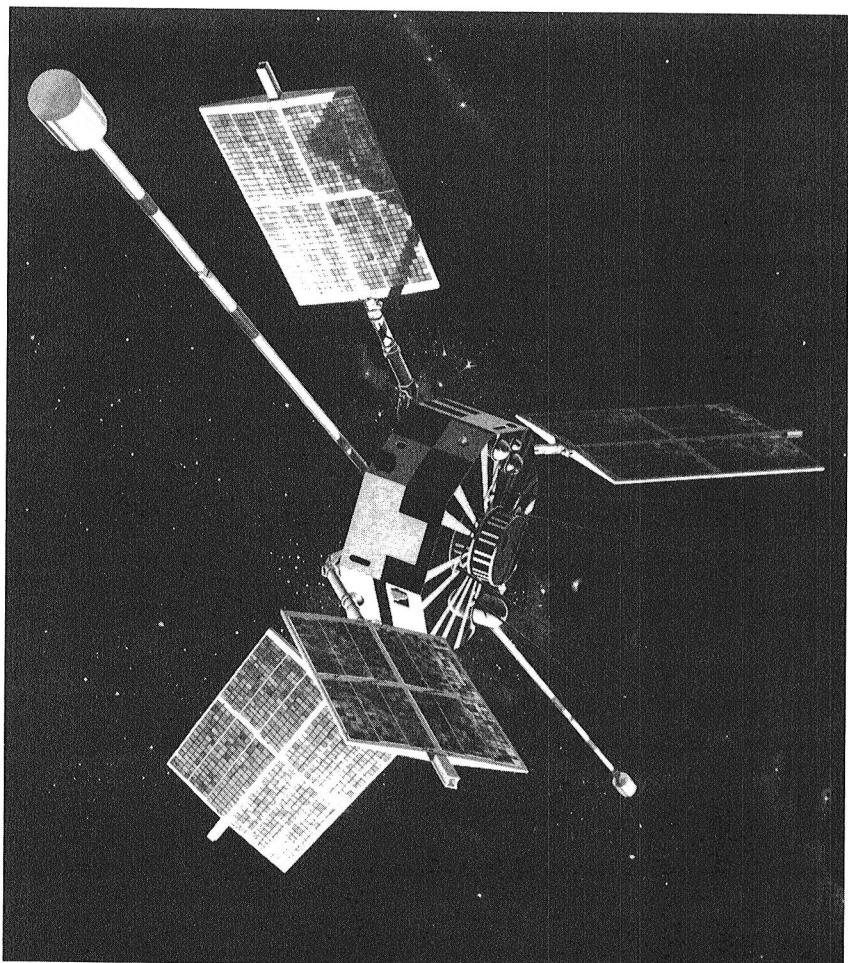


Figure 4. The IMP F Spacecraft in Flight Configuration.

The fourth group in the series consisted of only one spacecraft, the IMP I (Figure 5) which was placed in an eccentric orbit to further the scientific understanding of solar-terrestrial physics. Major advances in the lift capability of the launch vehicle [from 62-kilogram (kg) payload to nearly 300 kg without a kick motor on the spacecraft] allowed the spacecraft to carry larger and more complex experiment packages. Thus, on IMP I, two new experiments involving radio astronomy and electrical field measurements were added to the payload. These added significant technical complexity to the spacecraft while enhancing the scientific data return.

THE IMP PROGRAM—AN OVERVIEW

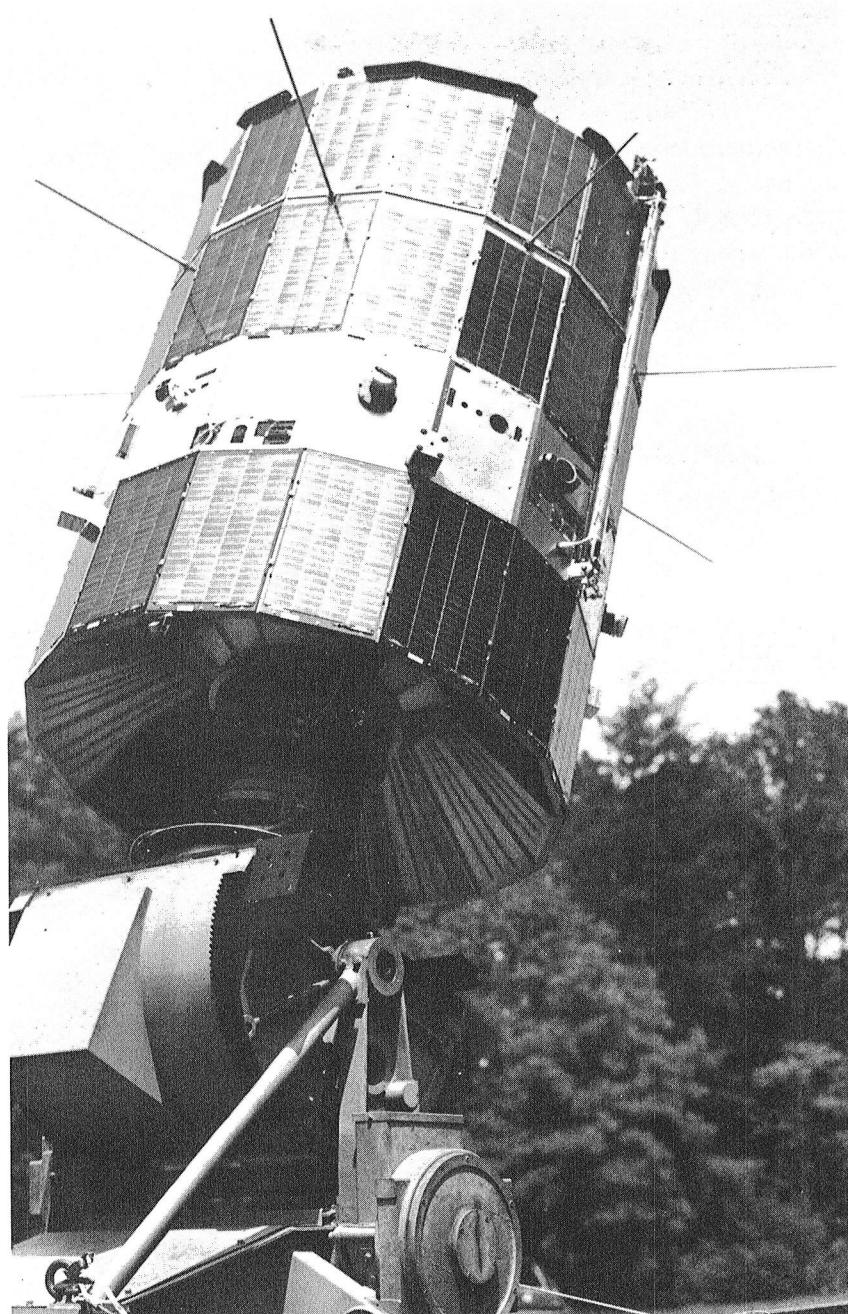


Figure 5. The IMP I Spacecraft is Shown on Spin Table in Readiness for Sun Spin Tests.

The fifth and last group in the series were designated IMPs H and J (Figure 6). These spacecraft were placed in a circular orbit halfway to the Moon. Their launch times were selected so that one spacecraft was on the Sun side of the Earth while the other was on the dark side initially. The experiments were chosen to provide correlative measurements of the Earth's magnetic tail and the solar wind.

IMPs A through E had many similar characteristics. Each weighed about 62 kg and used about 38 watts average power which was provided by solar cells and chemical batteries. Each had apogee kick motors to put them in proper orbit. There were some slight differences among the individual spacecraft. Exclusive of appendages such as antennas and

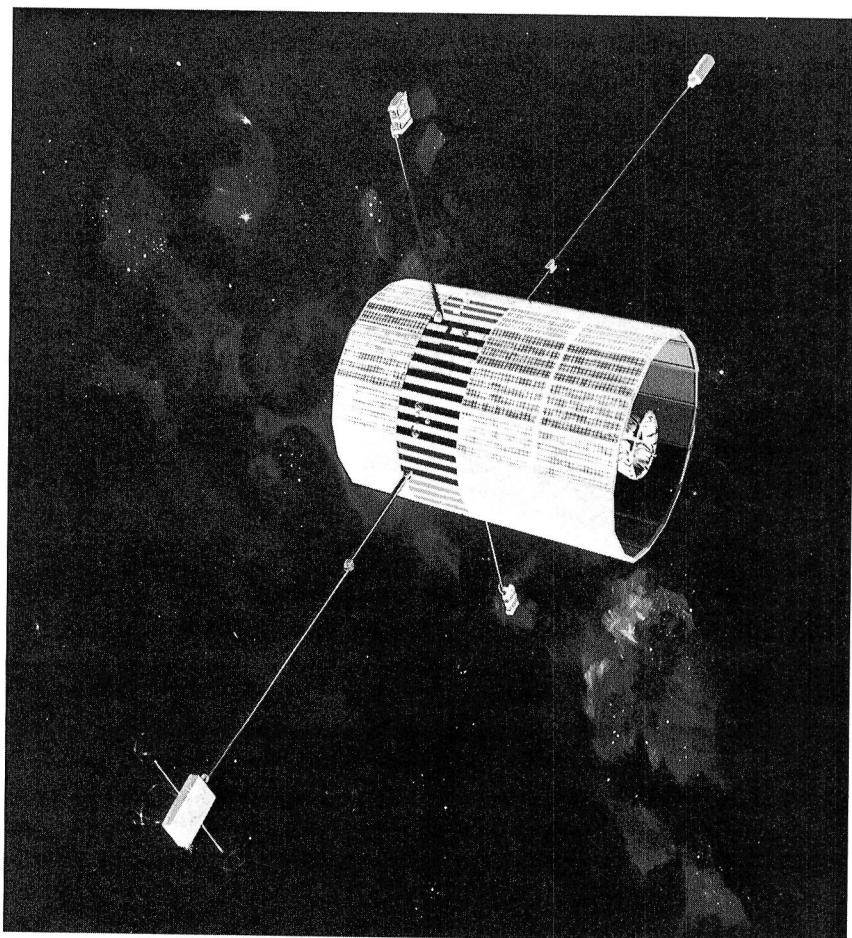


Figure 6. IMP H and J Spacecraft in Flight Configuration.

experiment booms, each spacecraft was shaped as an octagon drum measuring 71 centimeters (cm) across. IMPs A through C were about 30.5 cm high, while IMPs D and E were only 18 cm high. Each spacecraft, except IMP D, carried seven experiments; IMP D carried six experiments.

IMPs F and G were more complex than IMPs A through E, carrying 11 and 12 experiments, respectively. Their configurations and power consumptions were similar to those of IMPs A through C. They weighed 74 and 80 kg, respectively.

IMPs I, H and J represented the most significant increase in complexity in the IMP series. Each spacecraft was a 16-sided drum, measuring 135 cm across and 183 cm high. They weighed about 278 kg each, used about 110 watts average power and had a high data rate capability. Apogee kick motors were needed for these two spacecraft. They carried 12 experiments.

MISSION EXPERIMENTS

The scientific experiments carried into space by the IMP spacecraft measured the interaction of the Sun and Earth magnetic fields, electric fields (IMPs I, H, and J only), plasma, and energetic particles. Other experiments included the IMP E Micrometeorite Experiment, the IMP I, H and J Radio Astronomy Experiments, and the IMP D and E Passive Lunar Experiments.

Data gathered by the IMP spacecraft were used in support of the Apollo program. Radiation in the regions of space to be covered by the Apollo missions were studied by direct measurements prior to the launch of the first Apollo spacecraft. During the actual Apollo flights, these regions were monitored continuously by the IMP spacecraft and data describing radiation levels, particularly solar proton flux, were sent to Mission Control at Houston at 4-hour intervals. This near-real-time reporting was necessary to detect and monitor solar flare activity that might occur during the manned flights.

The experiments carried on each IMP spacecraft are listed in Table 2. Table 3 contains a complete list of experiments on IMPs A through J grouped by phenomena measured. In both tables, two-character codes have been assigned to individual experiments. The first character (alphanumeric) is the letter designator of the IMP spacecraft on which the experiment was flown. The second character (numeric) of the code has been sequenced, within each phenomena-measured grouping for each spacecraft, alphabetically by the principal investigator's name.

Table 2
List of IMP Spacecraft Experiments

IMP A	IMP C	IMP G
A1 Magnetometer	G1 Magnetometer	
A2 MIT Faraday Cup	G2 Electrostatic Analyzer	
A3 GSFC Faraday Cup	G3 Channeltron	
A4 Electrostatic Analyzer	G4 Ion Chamber and GM Tubes	
A5 Ion Chamber and GM Counters	G5 Solar Proton Monitor	
A6 Scintillator and GM Telescopes	G6 Low Energy Solid-State Telescope	
A7 Solid-State Telescope	G7 Low Energy Proton and Electron Differential Energy Analyzer	
IMP B	G8 Low Energy Proton Differential Energy Analyzer	
B1 Magnetometer	G9 Cosmic Ray Anisotropy	
B2 MIT Faraday Cup	G10 Solid-State Detectors	
B3 GSFC Faraday Cup	G11 Scintillator Telescope	
B4 Electrostatic Analyzer	G12 Solid-State Telescope	
B5 Ion Chamber and GM Counters		
B6 Scintillator and GM Telescopes		
B7 Solid-State Telescope		
IMP C	IMP I	
C1 Magnetometer	I1 AC Electric and Magnetic Fields	
C2 MIT Faraday Cup	I2 Fluxgate Magnetometer	
C3 GSFC Faraday Cup	I3 DC Electric Fields	
C4 Electrostatic Analyzer	I4 LASL Electrostatic Analyzer	
C5 Ion Chamber and GM Counters	I5 GSFC Electrostatic Analyzer	
C6 Scintillator and GM Telescopes	I6 Solid-State and GM Detectors	
C7 Solid-State Telescope	I7 Solar Proton Monitor	
IMP D	I8 Crystal Scintillator	
D1 GSFC Magnetometer	I9 Low Energy Proton and Electron Differential Energy Analyzer	
D2 Ames Magnetometer	I10 Solid-State and Scintillator Telescope	
D3 MIT Faraday Cup	I11 Solid-State Telescope	
D4 GSFC Faraday Cup	I12 Radio Astronomy	
D5 Ion Chamber and GM Counters		
D6 Solid-State and GM Counters		
D7 Selenodetic Studies		
D8 Cislunar RF Beacon		
IMP E	IMP H	
E1 GSFC Magnetometer	H1 Fluxgate Magnetometer	
E2 Ames Magnetometer	H2 Electric and Magnetic Fields	
E3 MIT Faraday Cup	H3 LASL Electrostatic Analyzer	
E4 GSFC Faraday Cup	H4 Faraday Cup	
E5 Ion Chamber and GM Counters	H5 GSFC Electrostatic Analyzer	
E6 Solid-State and GM Counters	H6 Crystal Scintillator	
E7 Micrometeorite Flux	H7 Low Energy Proton and Electron Differential Energy Analyzer	
E8 Selenodetic Studies	H8 University of Maryland Solid-State Detectors	
E9 Cislunar RF Beacon	H9 Solid-State and GM Detectors	
IMP F	H10 Solid-State and Scintillator Telescopes	
F1 Magnetometer	H11 Chicago Solid-State Telescopes	
F2 TRW Electrostatic Analyzer	H12 C.I.T. Solid-State Telescope	
F3 GSFC Electrostatic Analyzer	H13 NOAA Solid-State Detectors	
F4 Ion Chamber and GM Counters		
F5 Solar Proton Monitor		
F6 Low Energy Solid-State Telescope		
F7 Low Energy Proton and Electron Differential Energy Analyzer		
F8 Cosmic Ray Anisotropy	IMP J	
F9 GSFC Solid-State Telescope	J1 AC Electric and Magnetic Fields	
F10 Scintillator Telescope	J2 Fluxgate Magnetometer	
F11 Chicago Solid-State Telescope	J3 DC Electric Fields	
	J4 Electrostatic Analyzer	
	J5 Faraday Cup	
	J6 Low Energy Proton and Electron Differential Energy Analyzer	
	J7 Solid-State Detectors	
	J8 Solid-State and GM Detectors	
	J9 Solid-State and Scintillator Telescopes	
	J10 Chicago Solid-State Telescopes	
	J11 C.I.T. Solid-State Telescope	
	J12 NOAA Solid-State Detectors	

THE IMP PROGRAM—AN OVERVIEW

Table 3
IMP Experiments Grouped by Phenomena Measured

Experiment Code	Principal Investigator	Investigator Affiliation
-----------------	------------------------	--------------------------

Magnetic Field Experiments

A1	N.F. Ness	NASA/Goddard Space Flight Center
B1	N.F. Ness	NASA/Goddard Space Flight Center
C1	N.F. Ness	NASA/Goddard Space Flight Center
D1	N.F. Ness	NASA/Goddard Space Flight Center
D2	C.P. Sonett	NASA/Ames Research Center
E1	N.F. Ness	NASA/Goddard Space Flight Center
E2	C.P. Sonett	NASA/Ames Research Center
F1	N.F. Ness	NASA/Goddard Space Flight Center
G1	N.F. Ness	NASA/Goddard Space Flight Center
I1*	D.A. Gurnett	University of Iowa
	P.J. Kellogg	University of Minnesota
	T.L. Aggson	NASA/Goddard Space Flight Center
I2	N.F. Ness	NASA/Goddard Space Flight Center
H1	N.F. Ness	NASA/Goddard Space Flight Center
H2*	F.L. Scarf	TRW Systems Group
J1*	D.A. Gurnett	University of Iowa
	P.J. Kellogg	University of Minnesota
	T.L. Aggson	NASA/Goddard Space Flight Center
J2	N.F. Ness	NASA/Goddard Space Flight Center

Electric Field Measurements

I3	T.L. Aggson	NASA/Goddard Space Flight Center
I1*	D.A. Gurnett	University of Iowa
	P.J. Kellogg	University of Minnesota
	T.L. Aggson	NASA/Goddard Space Flight Center
H2*	F.L. Scarf	TRW Systems Group
J3	T.L. Aggson	NASA/Goddard Space Flight Center
J1*	D.A. Gurnett	University of Iowa
	P.J. Kellogg	University of Minnesota
	T.L. Aggson	NASA/Goddard Space Flight Center

*Plasma Experiments***

A2	H.S. Bridge	Massachusetts Institute of Technology
A3	G.P. Serbu	NASA/Goddard Space Flight Center
A4	J.H. Wolfe	NASA/Ames Research Center
B2	H.S. Bridge	Massachusetts Institute of Technology
B3	G.P. Serbu	NASA/Goddard Space Flight Center
B4	J.H. Wolfe	NASA/Ames Research Center

*Simultaneous ac electric and magnetic field measurements.

**University of Iowa LEPEDEA experiments on IMPs F, G, I, H, and J are listed under energetic particle experiments.

Table 3
IMP Experiments Grouped by Phenomena Measured (Continued)

Experiment Code	Principal Investigator	Investigator Affiliation
<i>Plasma Experiments (continued)</i>		
C2	H.S. Bridge	Massachusetts Institute of Technology
C3	G.P. Serbu	NASA/Goddard Space Flight Center
C4	J.H. Wolfe	NASA/Ames Research Center
D3	H.S. Bridge	Massachusetts Institute of Technology
D4	G.P. Serbu	NASA/Goddard Space Flight Center
E3	H.S. Bridge	Massachusetts Institute of Technology
E4	G.P. Serbu	NASA/Goddard Space Flight Center
F2	F.B. Harrison	TRW Systems Group
F3	K.W. Ogilvie	NASA/Goddard Space Flight Center
G2	K.W. Ogilvie	NASA/Goddard Space Flight Center
I4	S.J. Bame	Los Alamos Scientific Laboratory
I5	K.W. Ogilvie	NASA/Goddard Space Flight Center
H3	S.J. Bame	Los Alamos Scientific Laboratory
H4	H.S. Bridge	Massachusetts Institute of Technology
H5	K.W. Ogilvie	NASA/Goddard Space Flight Center
J4	S.J. Bame	Los Alamos Scientific Laboratory
J5	H.S. Bridge	Massachusetts Institute of Technology
<i>Energetic Particle Experiments</i>		
A5	K.A. Anderson	University of California at Berkeley
A6	F.B. McDonald	NASA/Goddard Space Flight Center
A7	J.A. Simpson	University of Chicago
B5	K.A. Anderson	University of California at Berkeley
B6	F.B. McDonald	NASA/Goddard Space Flight Center
B7	J.A. Simpson	University of Chicago
C5	K.A. Anderson	University of California at Berkeley
C6	F.B. McDonald	NASA/Goddard Space Flight Center
C7	J.A. Simpson	University of Chicago
D5	K.A. Anderson	University of California at Berkeley
D6	J.A. Van Allen	University of Iowa
E5	K.A. Anderson	University of California at Berkeley
E6	J.A. Van Allen	University of Iowa
F4	K.A. Anderson	University of California at Berkeley
F5	C.O. Bostrom	Johns Hopkins University/Applied Physics Laboratory
F6	W.L. Brown	Bell Telephone Laboratories
F7	J.A. Van Allen	University of Iowa
F8	K.G. McCracken	University of Texas at Dallas
F9	F.B. McDonald	NASA/Goddard Space Flight Center
F10	F.B. McDonald	NASA/Goddard Space Flight Center
F11	J.A. Simpson	University of Chicago
G3	R.P. Lin	University of California at Berkeley
G4	K.A. Anderson	University of California at Berkeley
G5	C.O. Bostrom	Johns Hopkins University/Applied Physics Laboratory

Table 3
IMP Experiments Grouped by Phenomena Measured (Continued)

Experiment Code	Principal Investigator	Investigator Affiliation
<i>Energetic Particle Experiments (continued)</i>		
G6	W.L. Brown	Bell Telephone Laboratories
G7	J.A. Van Allen	University of Iowa
G8	L.A. Frank	University of Iowa
G9	K.G. McCracken	University of Texas at Dallas
G10	F.B. McDonald	NASA/Goddard Space Flight Center
G11	F.B. McDonald	NASA/Goddard Space Flight Center
G12	J.A. Simpson	University of Chicago
I6	K.A. Anderson	University of California at Berkeley
I7	C.O. Bostrom	Johns Hopkins University/Applied Physics Laboratory
I8	T.L. Cline	NASA/Goddard Space Flight Center
I9	L.A. Frank	University of Iowa
I10	F.B. McDonald	NASA/Goddard Space Flight Center
I11	J.A. Simpson	University of Chicago
H6	T.L. Cline	NASA/Goddard Space Flight Center
H7	L.A. Frank	University of Iowa
H8	G. Gloeckler	University of Maryland
H9	S.M. Krimigis	Johns Hopkins University/Applied Physics Laboratory
H10	F.B. McDonald	NASA/Goddard Space Flight Center
H11	J.A. Simpson	University of Chicago
H12	E.C. Stone	California Institute of Technology
H13	D.J. Williams	NOAA/Space Environment Laboratory
J6	L.A. Frank	University of Iowa
J7	G. Gloeckler	University of Maryland
J8	S.M. Krimigis	Johns Hopkins University/Applied Physics Laboratory
J9	F.B. McDonald	NASA/Goddard Space Flight Center
J10	J.A. Simpson	University of Chicago
J11	E.C. Stone	California Institute of Technology
J12	D.J. Williams	NOAA/Space Environment Laboratory

Other Experiments

Selenodesy		
D7*	W.M. Kaula	University of California at Los Angeles
E8*	W.M. Kaula	University of California at Los Angeles
RF Beacon		
D8*	A.M. Peterson	Stanford University
E9*	A.M. Peterson	Stanford University
Micrometeorites		
E7	J.L. Bohn	Temple University
Radio Astronomy		
I12	W.C. Erickson	University of Maryland

*Passive experiments.

Chapter 2

IMP Development

The development and successful launch and operation of a major spacecraft requires not only a lot of resources, but a lot of dedication and hard work from the hundreds of people involved. The Interplanetary Monitoring Platform Satellites Program was no exception. It brought together managers, engineers, scientists, technicians, and support personnel from many Government organizations, several universities from across the country and a number of industrial contractors.

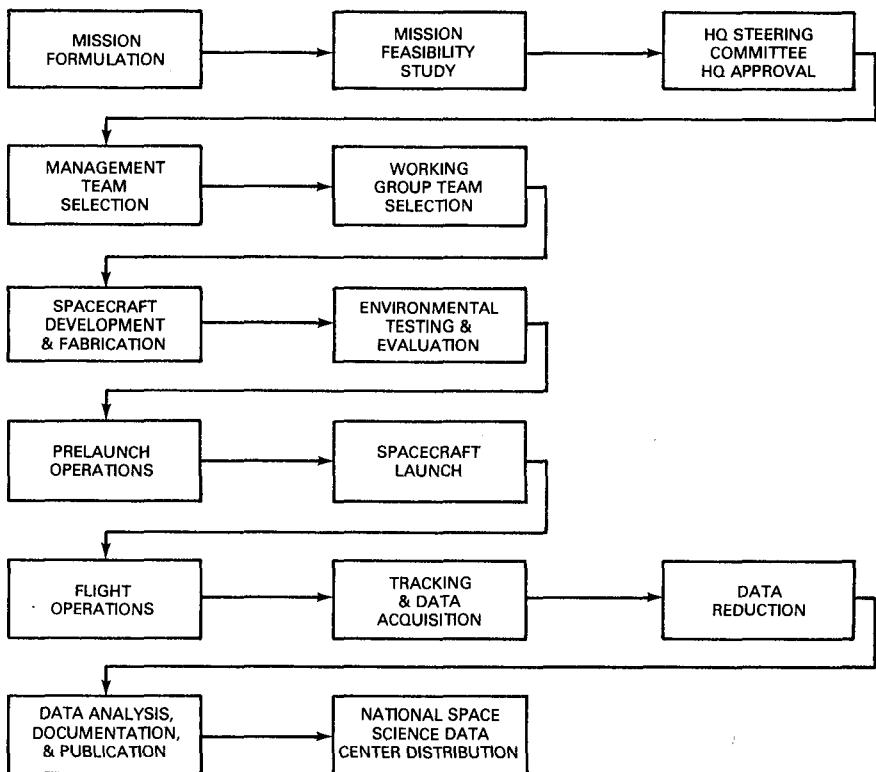
The key to the success of a joint or coordinated effort such as the IMP project is in the ability to effectively manage the resources and talents which are brought together to meet a common objective. Management and development responsibility for the IMP satellite project was assigned from the outset to NASA's Goddard Space Flight Center.

The development of satellites at GSFC was not new. By 1961, when the IMP project was assigned to GSFC, the space center had already developed a number of satellites which had been highly successful. In order to accomplish this, GSFC already had an established spacecraft development policy and procedures which could be followed in implementing the IMP project.

THE DEVELOPMENT PROCESS

The development of a spacecraft from the time the mission is formulated to the time the data is being retrieved from the satellite is a highly complex, multi-faceted series of activities which must be managed in an orderly and timely manner. The sequence of activities or events for the IMP project followed the development steps shown in the flow chart on the following page.

The first three boxes give the required preliminary activities for all NASA space missions. One of the first tasks is to formulate a detailed Project Plan which outlines the functional requirements of the spacecraft and experiments and their inter-relationships necessary to obtain the desired scientific data. In the IMP project, a Project Plan was pre-



pared for each group of spacecraft. Following the approval of the mission by NASA Headquarters, the organizational activities began with the selection of the management team which is responsible for implementing the project. Assignment of the IMP project to GSFC was followed by the selection of an IMP Project Manager and an IMP Project Scientist. In a major spacecraft project, the Project Manager is the senior NASA representative responsible for assuring the performance of all functions necessary for management of the project. These include project-wide planning and evaluation, systems integration, systems engineering, scheduling, budgetary and financial planning, technical monitoring of contracts, and project reporting.

Another member of the Project Office is the Project Scientist. This individual is responsible for the scientific aspects of the mission and its experiments and for obtaining the maximum experimental data consistent with the objectives of the project. The scientist also is responsible for optimizing the weight, power, space, and telemetry-time distributions

among the experiments as well as the location of all experiments in the spacecraft and the interface between each experiment and other experiments or spacecraft components.

Other key personnel, principally subsystem engineers in various technical specialties, are an integral part of the project management team. A Working Group Team (WGT) is also formed which is normally comprised of key personnel from the various technical support groups responsible for a particular development functional area. The major functional areas in the development process are concerned primarily with the spacecraft system design, power systems, data handling, thermal, stabilization and control, integration and testing and reliability and quality control.

In the IMP project, it was desired to develop the IMP spacecraft in groups (i.e., IMPs A-B-C, D-E, F-G, I, and H-J) so that one operational satellite would be continuously in orbit, monitoring and mapping the radiation environment during all phases of an 11-year solar cycle. Each group of satellites were to be developed on a "quick-turnaround" launch basis so that experience gained on a preceding flight could be used to upgrade and refine experiments planned for later spacecraft.

PHASED PROJECT DEVELOPMENT

The management approach used by GSFC to implement the IMP development process involved a series of activities called "phased project planning and development". This concept was based on the sequential performance of the following phases of effort:

- 1) Feasibility Study (Phase I)
- 2) Conceptual Design (Phase II)
- 3) Engineering Test Unit Fabrication (Phase III)
- 4) Proto-Flight Unit Fabrication and Tests (Phase IV)
- 5) Flight Unit Fabrication (Phase V)
- 6) Launch Site Operations (Phase VI)

The following paragraphs discuss the phased project planning and development activities used in the IMP project.

Phase I-Feasibility Study

The first task of the IMP project was to investigate the feasibility of combining all the proposed experiments into a single integrated system design. In addition to the feasibility study managers responsible for specific technical areas of support also conducted studies and provided

information to the project. This study considered the engineering aspects such as spacecraft mechanical and electrical design, thermal, power, data handling, stabilization and control, fabrication, assembly, electrical and mechanical integration, ground support, environmental and spacecraft reliability and quality control testing, flight data acquisition, orbital criteria, launch vehicle and launch operation, and pre- and post-launch activities.

Phase II—Conceptual Design

The preliminary or conceptual design phase follows the feasibility study phase in the phased development concept. In this phase, designs are prepared for mechanical, power, thermal, data handling and communications elements of the spacecraft.

In preparing preliminary and conceptual mechanical designs and configurations for the IMP satellites, the GSFC mechanical engineering group considered such problems as vehicle compatibility; spacecraft sizes, shapes and weights; balance and spin stabilization factors; experiment requirements; thermal loads; and the overall mission requirements. The designs and layouts enabled the project engineers to pinpoint, at an early stage, incompatibilities between experimental requirements and what could actually be accomplished. The engineers designed redundant systems when possible to assure success of a particular function. There was essentially no redundancy in the early series because of severe weight constraints.

During this phase of the spacecraft development process, preliminary electronic designs also were prepared. By knowing where experiments would be located within the body or on the surface of the spacecraft, GSFC engineers could design the wiring harness required for the interconnection of experiments, sensors, power sources, and radio frequency (RF) equipment. Engineers also began to design the ground system needed for integrating and performing operational tests of the spacecraft during environmental testing and pre-launch calibrations. A feasibility study also was conducted for spacecraft thermal control. This study determined the need for either active or passive temperature control, coating materials, and the probable patterns and coatings required to obtain desired temperature levels at the various locations where instrumentation or experiments were temperature-sensitive.

In parallel with this effort, the project engineer responsible for the spacecraft electrical power supply prepared preliminary designs including the numbers and types of solar cells, size and type of batteries, power

conditioning and controlling equipment (converters, regulators, charge control and undervoltage and fusing required). The engineer responsible for spacecraft antennas and communications also prepared preliminary drawings and designs for the antennas which resulted in a mock-up of the spacecraft antenna system called the antenna model (AM). The mock-up was used to study possible RF interference problems and to determine the antenna impedance match and radiation patterns of the antennas.

Phase III-Engineering Test Unit (ETU) Fabrication

After all computations and design drawings were completed and accepted, an ETU was fabricated and assembled. The primary purpose of this unit was to provide a model which could be used to thoroughly test the satellite's structural integrity before ordering hardware and materials to be used in the proto-flight and flight units. To accomplish its intended function, the ETU had to be structurally representative of the flight unit. The weight of the ETU had to be the same as, or slightly more than, that of the flight unit, and the ETU had to have essentially the same internal weight distribution.

The method used to achieve the proper weight with the ETU had been used on earlier spacecraft projects. It involved fabrication of wooden blocks duplicating the sizes and weights of the actual experiment or associated electronic packages. These blocks were weighted with lead slugs placed internally at their centers of gravity. The weighted blocks were fastened to predetermined locations in the IMP test unit, and the unit was then tested for structural integrity using prototype level loads and stresses, which were more severe than those to be encountered during flight. The locations of the actual experiment instrument packages were shifted if the test results using the blocks indicated potential structural integrity problems.

After the ETU was assembled with its weighted dummy components, a series of sinusoidal and random vibration tests in the thrust and lateral axes were performed. If any failures were detected during or upon completion of any test, the failed component was redesigned and re-tested before proceeding with other tests. The ETU was then subjected to acceleration, spin and appendage erection tests. If any component failure occurred during any of these tests, the component was redesigned and retested until its design proved satisfactory.

After the ETU was subjected to de-spin tests it was fitted with thermal models of each experiment and instrument and exposed to simulated

solar radiation to test the thermal control and determine internal temperature gradients.

Before the satellite design was completed, the task of designing, redesigning, and fabricating encapsulation molds for the experiments and subsystems was undertaken. The molds prevented the experiment and subsystem frames or containers from buckling during the encapsulation process, thus ensuring that the subsystem mechanical interface requirements would be met.

Phase IV—Proto-Flight (PF) Unit Fabrication and Tests

The IMP project management altered the customary practice of building a prototype unit in an attempt to get the spacecraft built and launched in the shortest time possible and to save costs. In this project, the prototype unit was fabricated with “flight-qualified” parts and was designated the “proto-flight” (PF) unit. It was the first of the series to be built, was tested at prototype test levels, and was the first of the series to be launched.

In the development process, this proto-flight fabrication phase was the first time that all the experiments and subsystems were mechanically integrated with the spacecraft structure. The first step in fabricating the IMP spacecraft began with the assembly of the main structure. This was usually a joint effort by the structural engineers and electronic integration team so that the wiring harness could be installed early to avoid damaging the harness. The subsystems and electronic components were installed, after being inspected, measured, and weighted (Figures 7 and 8). Examinations were conducted for hole alignments, proper connector mating, freedom from mechanical interference between components and structure, and proper seating. When the spacecraft was completely assembled, it was released to the electronic integration team for electronic checkout.

The mechanical and electronic integration of the subsystems into the structure occasionally revealed problems such as electromagnetic or RF interference of an experiment or its electronic circuitry with another experiment, or RF leakage into other circuitry. These problems often required additional shielding, filtering, or rerouting of the wiring. These changes were mostly minor, although some subsystems had to be redesigned, thus causing a structural design change. As a result, it was desirable to delay production of the flight unit until the proto-flight unit had undergone nearly complete environmental testing.

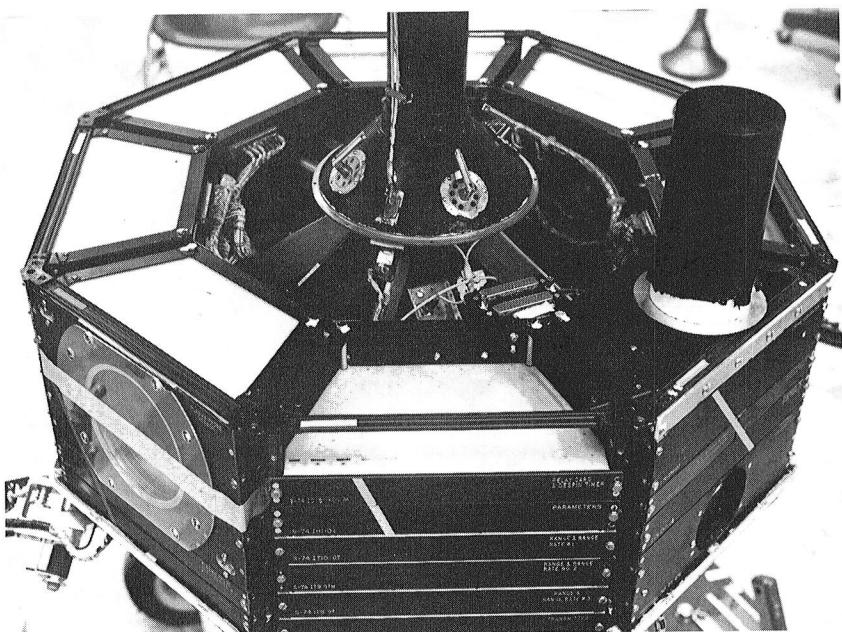


Figure 7. Experiment Packages and Associated Electronics Installed into Spacecraft Structure.

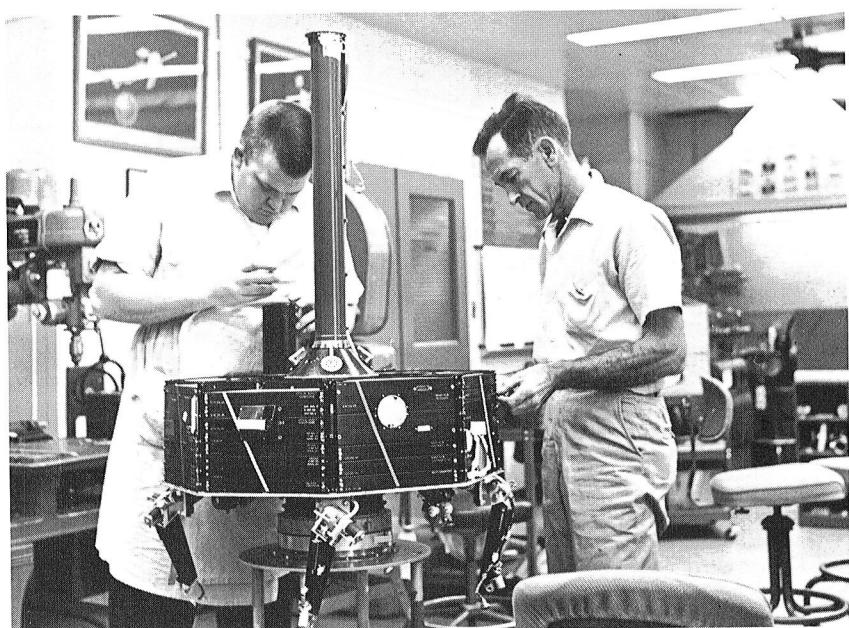


Figure 8. Technicians Perform Final Mechanical Assembly of Experiment Boom.

On IMPs A, B, and C, and AIMPs D and E, encapsulation and conformal-coating of electronic and other components was performed after completion of electronic integration. This procedure allowed the instruments and experiments to be functionally integrated in the system and operated in all flight modes. Once the proto-flight unit was functioning properly, the instruments and experiments were removed, conformally-coated, encapsulated, and reassembled into the proto-flight unit to determine if the encapsulant changed the characteristics of the experiments and subsystems.

On IMPs F, G, I, H, and J, the electronic instruments were pretested as subsystems, then encapsulated prior to electronic integration. This proved to be an efficient method since prior experience, better interface definition and control minimized the risk of system incompatibility that would require redesign or change to the instruments.

The proto-flight unit was subjected to a series of environmental tests which are summarized in the flow chart of Figure 9. First, the unit was attached to a balance machine and a preliminary or rough balance was performed. The purpose of this test was to assure the spacecraft mass distribution was close to flight configuration. The next step was to subject the unit to vibration levels fifty percent more than those anticipated in flight (Figure 10). This test was to assure that the operating electronic instruments and experiments could survive the launch vehicle environment.

Temperature tests were conducted to detect any defect or weakness caused by thermal loads on the electronic circuitry before performing the thermal-vacuum system tests.

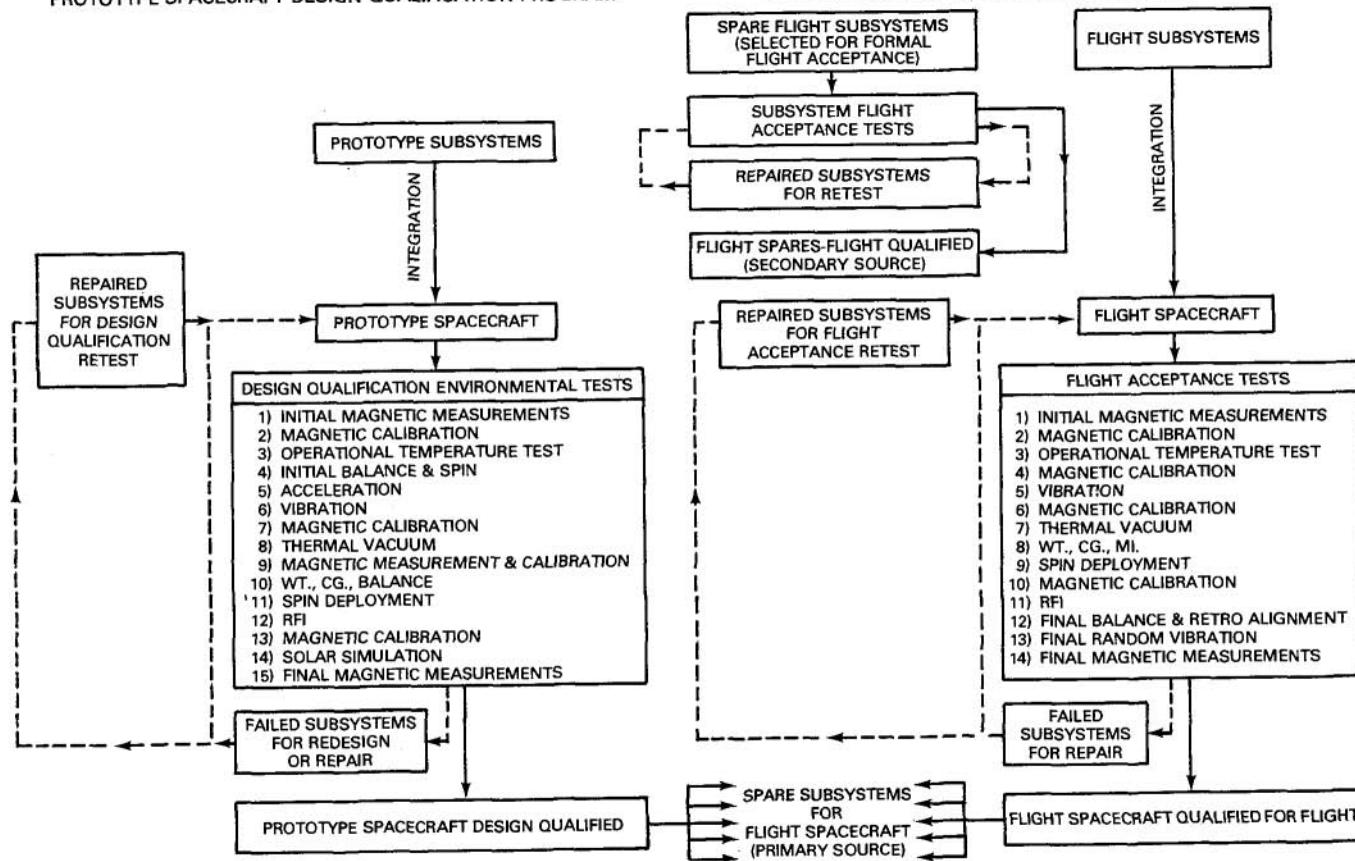
Thermal-vacuum tests also were conducted to determine the spacecraft's integrity when subjected to a simulated environment that duplicated most aspects of the actual environment that the spacecraft would experience in space.

Compatibility tests were conducted to determine the spacecraft's RF compatibility with the tracking stations and included a test for reception of erroneous commands. All command modulator tones were transmitted to the spacecraft, and the spacecraft was checked for proper reception and interpretation of commands.

In conjunction with this, an RF interference test was conducted with the flight unit set up in the launch configuration in a screen room. All ground support equipment (GSE) connections were made with the spacecraft in the pre-launch configuration. The purpose of the test was to check the spacecraft and GSE for transmission and reception of spurious RF signals.

PROTOTYPE SPACECRAFT DESIGN QUALIFICATION PROGRAM

FLIGHT SPACECRAFT ACCEPTANCE PROGRAM



IMP DEVELOPMENT

Figure 9. Environmental Test Program.

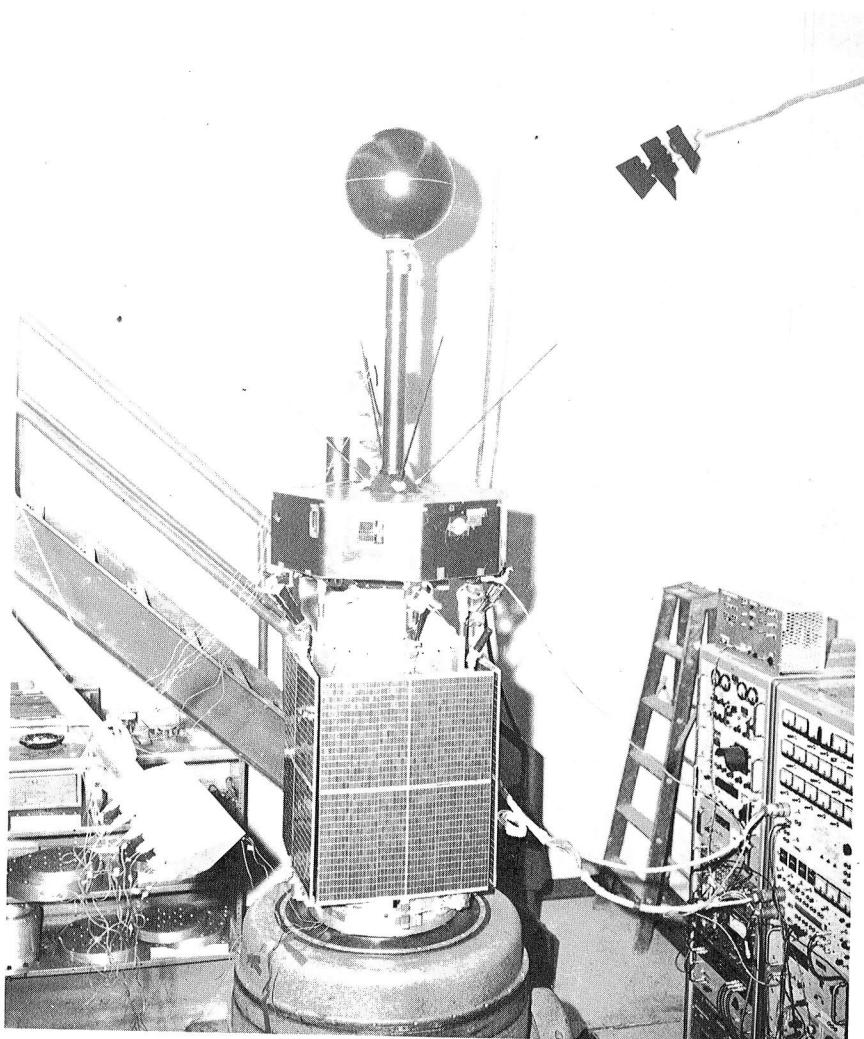


Figure 10. The IMP Spacecraft was Subjected to Proto-Flight Vibration Levels in the Test and Evaluation Facility at the Goddard Space Flight Center.

A Sun-spin test was conducted on all units using a gimbaled spin table at GSFC (Figure 11). These tests were designed to check the operation of the optical aspect system and the sensor's response to a real Sun with the spacecraft spinning. The solar paddle performance could not be evaluated during the Sun-spin test at either GSFC or at the launch site due to backscatter (reflections) from the surrounding environment.

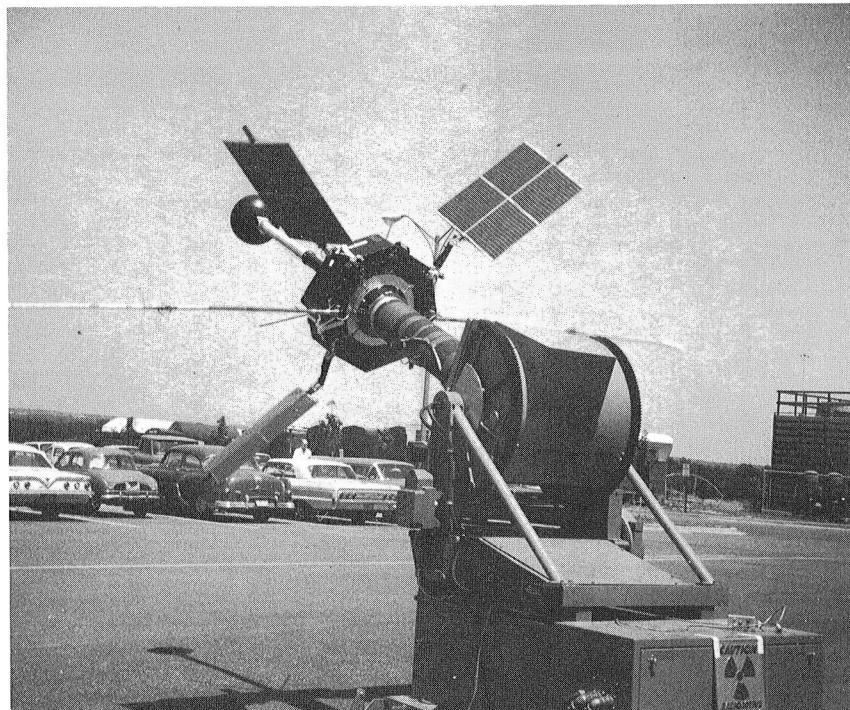


Figure 11. The Spacecraft was Subjected to Sun-Spin Tests to Calibrate the Solar Paddles and Measure Energy Outputs.

After all tests were completed, final balance of the unit was performed (Figure 12). Since the IMP spacecraft were spin stabilized, the static and dynamic balance were critical. Due to design of the structure and the location of the experiments and appendages, it was necessary to distribute the weights and retest to ensure the balance specifications were achieved. The spacecraft was then placed in a protective and controlled clean environment until it was prepared for shipment to the launch site.

As indicated earlier, the proto-flight spacecraft was designed as a prototype and tested at prototype levels, i.e., the time duration and the vibration levels exceed the expected flight levels. The PF spacecraft was then launched as IMP A and it became the first satellite of the series. Follow-on spacecraft of the same series were tested and evaluated at flight unit levels, (i.e., the IMP A proto-flight unit was tested at the prototype level and the IMP B, C flight units were tested at the flight unit level).

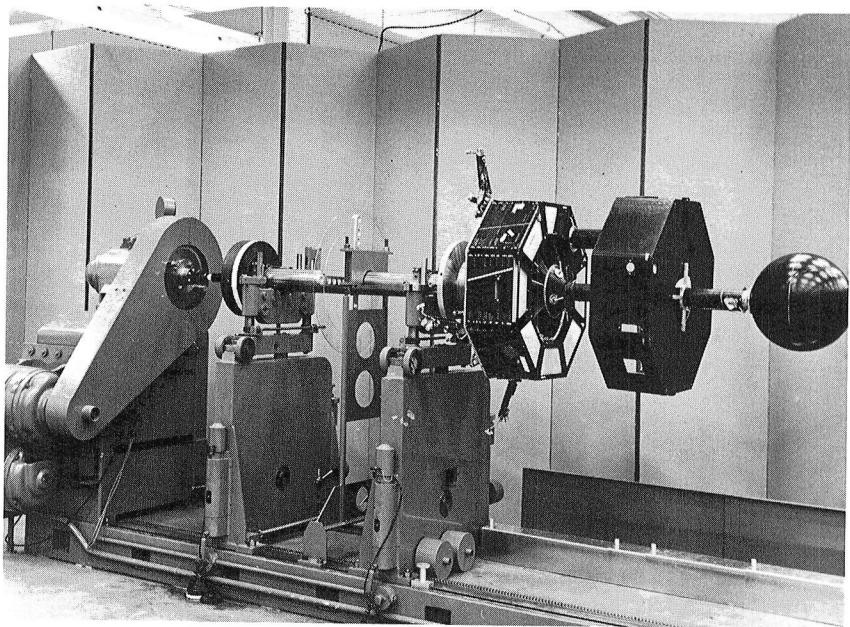


Figure 12. The IMP Spacecraft was Installed on a Balance Machine Where it Underwent Dynamic Balance. The Top Cover was Removed to Facilitate the Installation of Required Balance Weights.

Phase V-Flight Unit (FU) Fabrication

The second in the series to be built was the "Flight unit" and, as indicated earlier, the best time for assembling the FU was after the PF unit was environmentally tested. However, since some problems were encountered during PF unit integration and test, it became necessary to begin fabrication and assembly of the FU before PF qualification was completed due to schedule and cost considerations.

As the electronic state-of-the-art improved and the vehicle capability increased during the duration of the project, the physical sizes of sensors and support electronics increased and a larger number of experiments could be accommodated. Micro-miniaturization added more complexity to the experiment electronics but greatly enhanced the scientific yield. All of these improvements reflected increased effort in the design, fabrication, integration and testing of the spacecraft.

A passive thermal control system involving the use of paint patterns was not finalized until the solar environmental simulation tests were completed. Based on these results, the final paint patterns were deter-

mined and were sometimes applied at the launch site. The reason for this was the short time left between the end of thermal vacuum testing and field operations. This was a joint effort by the mechanical project engineer and the thermal project engineer to complete this phase before the spacecraft was mated to the vehicle.

Figure 9 gives the environmental test and other tests performed on all flight units of the IMP satellite series before launching. Many of the tests described in the preceding section were performed on the FU (see Figure 13). However, one of the reasons for testing at proto-flight levels was to detect significant design problems in the spacecraft as early as possible. Thus, the problems normally disclosed by environmentally testing the FU generally involved defects in workmanship, such as poor solder joints, faulty connectors, defective fasteners, and similar items.

After the environmental tests were completed and the flight unit was accepted, the spacecraft was shipped to the launch site in sealed aluminum containers with vibration-isolation mountings and a dessicant or dry-nitrogen atmosphere. Shipping and handling operations were closely observed to ensure that any damage to the spacecraft was detected and immediately corrected.

Phase VI—Launch Site Operations

The IMP satellites normally were shipped to the launch site three to five weeks prior to launch. The last week of this time was used for operations involving the launch vehicle and satellite together (Figure 14). Prior to this, the satellite underwent operational checks and calibration.

The first few days at the launch site were spent in checking the spacecraft's overall performance. The first complete spacecraft check in the field started in a clean room on the second day of testing. The ground support equipment was cleaned and decontaminated before placing the equipment in a clean room separate from that where the spacecraft resided. Cables were run from the spacecraft and the clean room anteroom.

All spacecraft and experiment systems were carefully checked and compared with previous test records. Sensors were checked against calibration sources, and the data obtained were compared with data previously obtained. When it was established that the unit was operating properly, it was transported to an antenna range for RF transmission

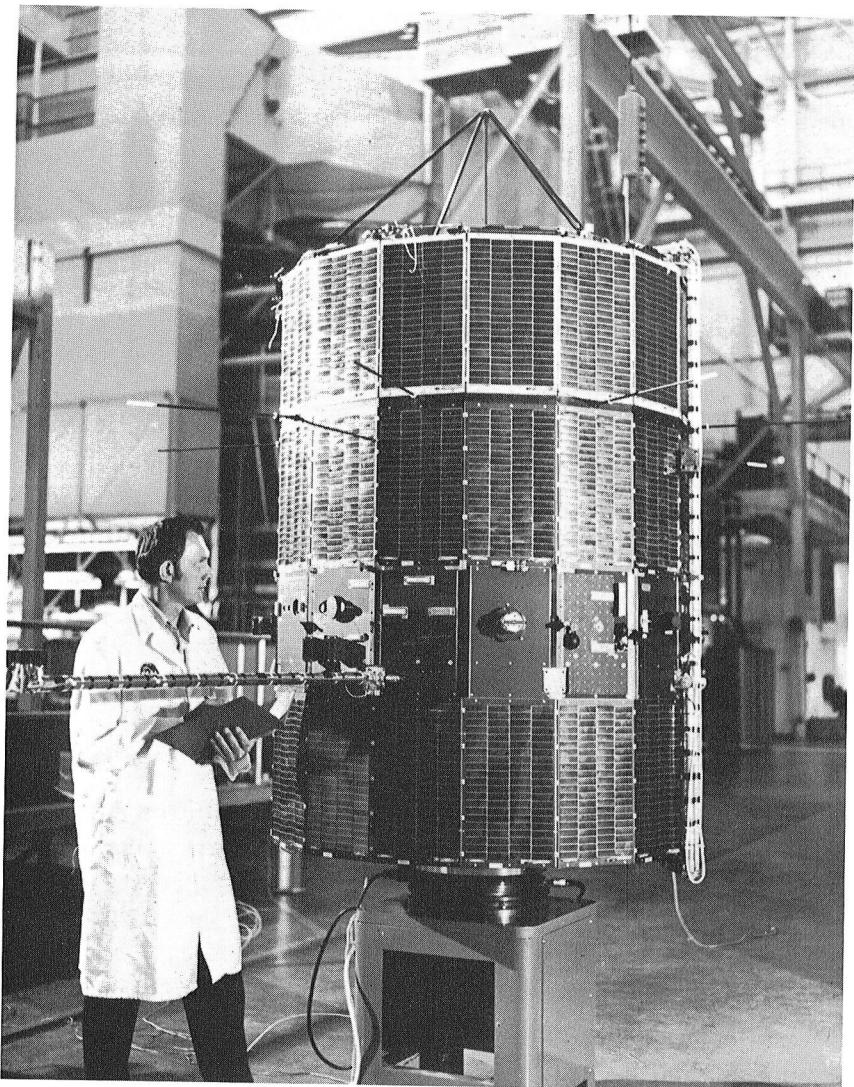


Figure 13. GSFC Technician Obtains Precision Measurements of the Final Center of Gravity Checks on the IMP I Flight Unit Spacecraft.

tests. After this, the unit was taken to the vehicle spin facility for mating with the last stage and for attaching supports to the last stage for the spacecraft booms, paddles and antennas. When this work was complete, the spacecraft and last stage were placed in containers and taken to the gantry, and the complete assembly was attached to the second stage of

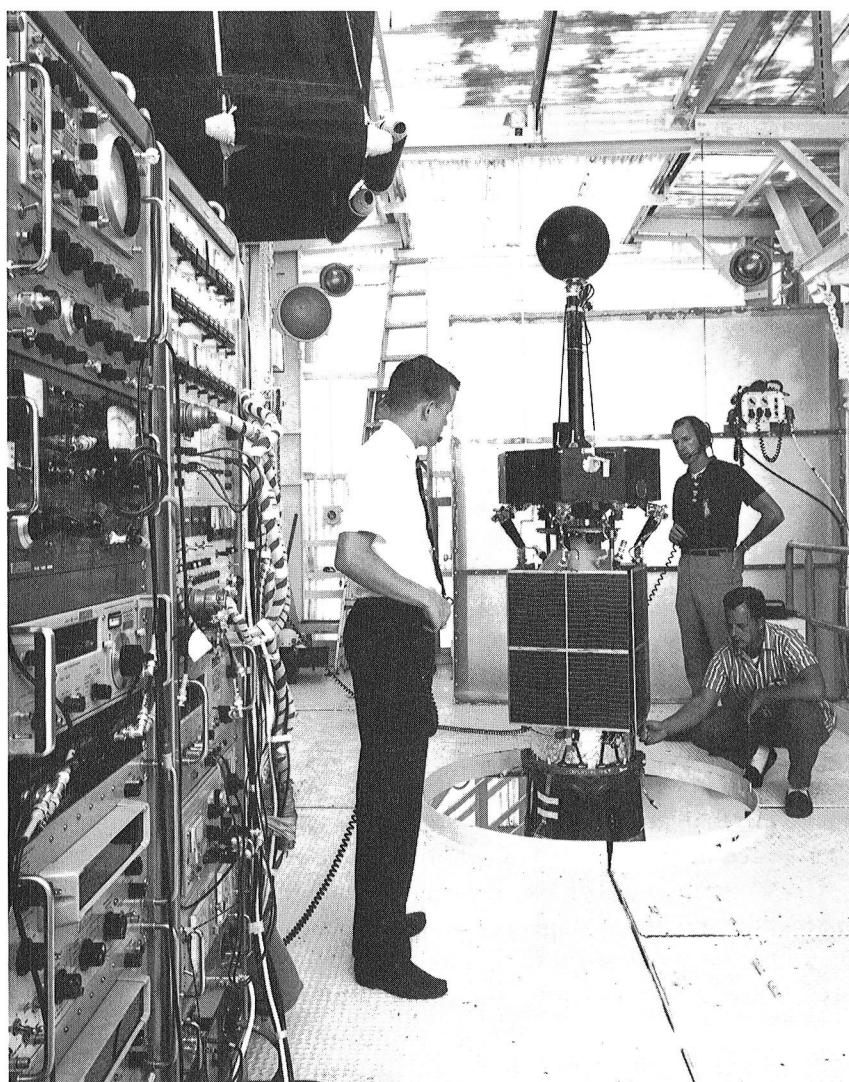


Figure 14. Spacecraft and Third Stage Vehicle were Mated as a Unit, and the Unit Secured to the Launch Vehicle.

the launch vehicle. IMPs A, B and C spacecraft were attached to the third stage and positioned to assure proper alignment of the spacecraft and the third stage axis; the complete assembly was then dynamically balanced. (Since the actual solar paddles on the flight unit were highly susceptible to damage, dummy paddles were used for this operation.)

On AIMPs D and E (Figure 15), a different method was employed for the balance operation. The spacecraft and third stage combination were mounted on a spin-balance table and tested to verify the GSFC spin-balance test results. This was the only balance check conducted. However, the fourth stage retro-motor was aligned with the spacecraft prior to the spacecraft's mating with the third stage.

IMPs F and G were dynamically balanced at GSFC. They were mated and aligned to the third stage at the launch site. The third stage had previously been spin-balanced at the launch site. A balance check was later made to ensure that the combination of spacecraft and third stage were properly balanced.

The IMP I spacecraft was dynamically balanced at GSFC prior to shipment to the launch site. The third stage was balanced at the site prior to mating the spacecraft with the third stage. Weights that were to be added to achieve the desired balance between the third stage and the spacecraft were determined analytically. The spacecraft and third stage were mated in the spin-balance facility and the combination was mounted on the spin table. An optical checkout system was used on the IMP I spacecraft for the first time to measure misalignment.

Dynamic balance of both the IMP H and J spacecraft was performed at GSFC. The live kick motors were balanced empirically at the NASA Wallops Island facility. At the launch site spin balance facility, the kick motors were carefully aligned in the spacecraft prior to assembling the spacecraft to the last stage of the launch vehicle. The entire assembly was then tested at the launch site spin balance facility.

Upon completion of the balance and alignment operations, the spacecraft and third stage assembly were placed in a container for handling the mated third stage and spacecraft combinations and delivered to the gantry for installation onto the second stage.

A final test to determine possible RF interference on the gantry was conducted by turning the spacecraft on, transmitting a signal, and receiving and analyzing this signal a mile away. The purpose of the test was to determine whether any spacecraft signals interfered with launch vehicle telemetry signals or other RF systems and determine the interference the vehicle system might have on the spacecraft system.

Additional spacecraft operational checks were made at this time (Figure 16). The fairing or nose cone was then installed over the spacecraft (Figures 17 and 18). After completing this installation and performing additional minor tasks, the IMP spacecraft could be launched.

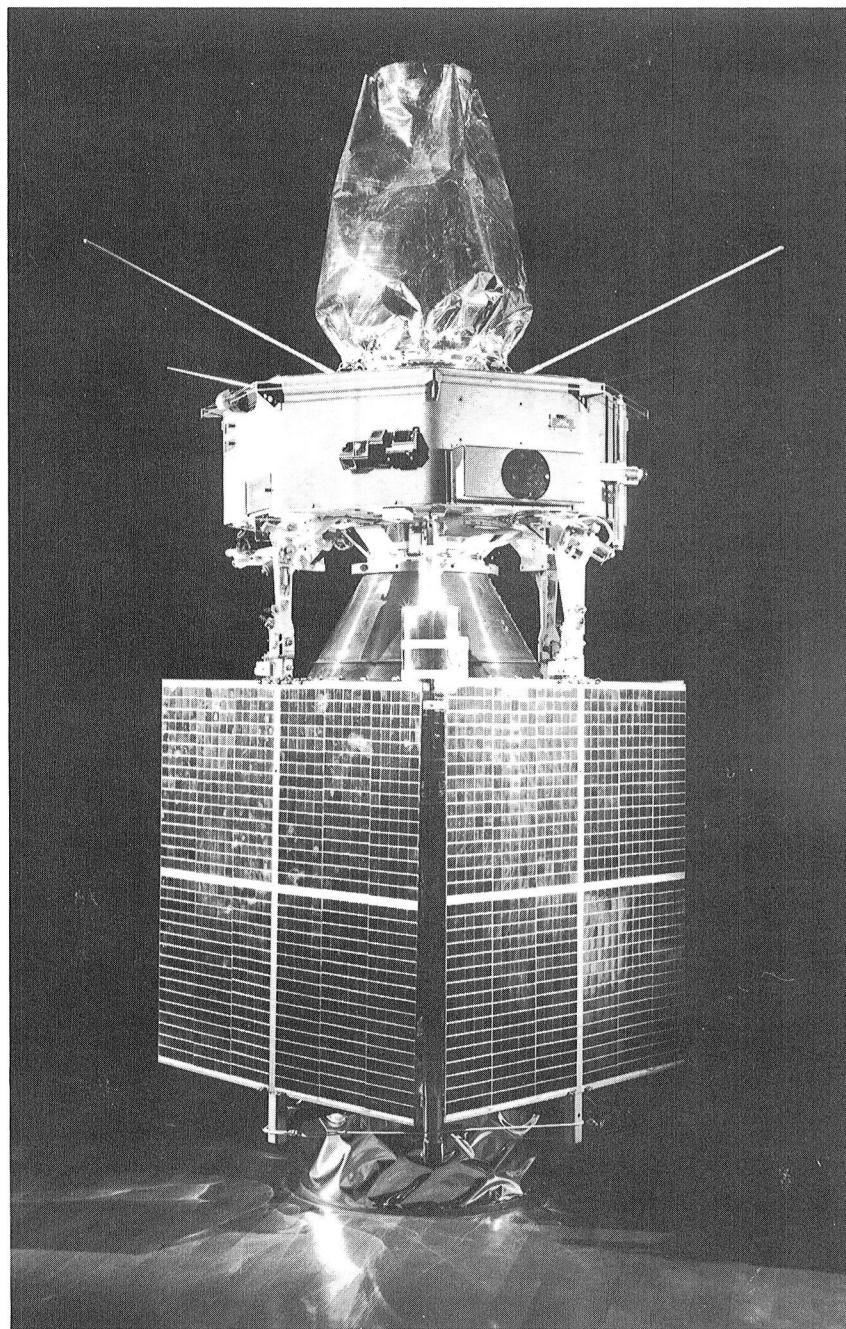


Figure 15. The AIM-P D Spacecraft in Launch Configuration, Assembled on Dummy Third Stage.

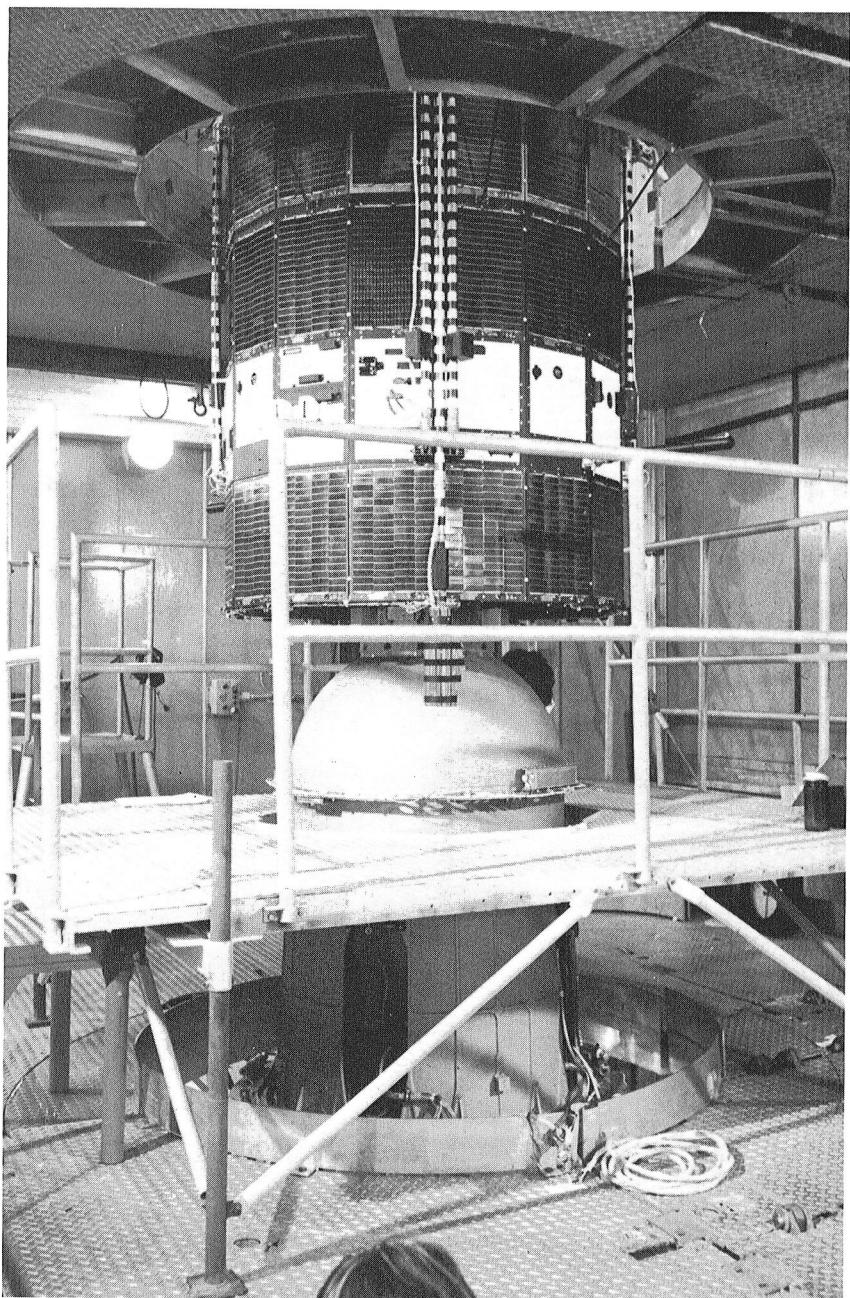


Figure 16. The IMP J Spacecraft on the Gantry was Mated with the Launch Vehicle. Final Flight Operation Checks were then Conducted.

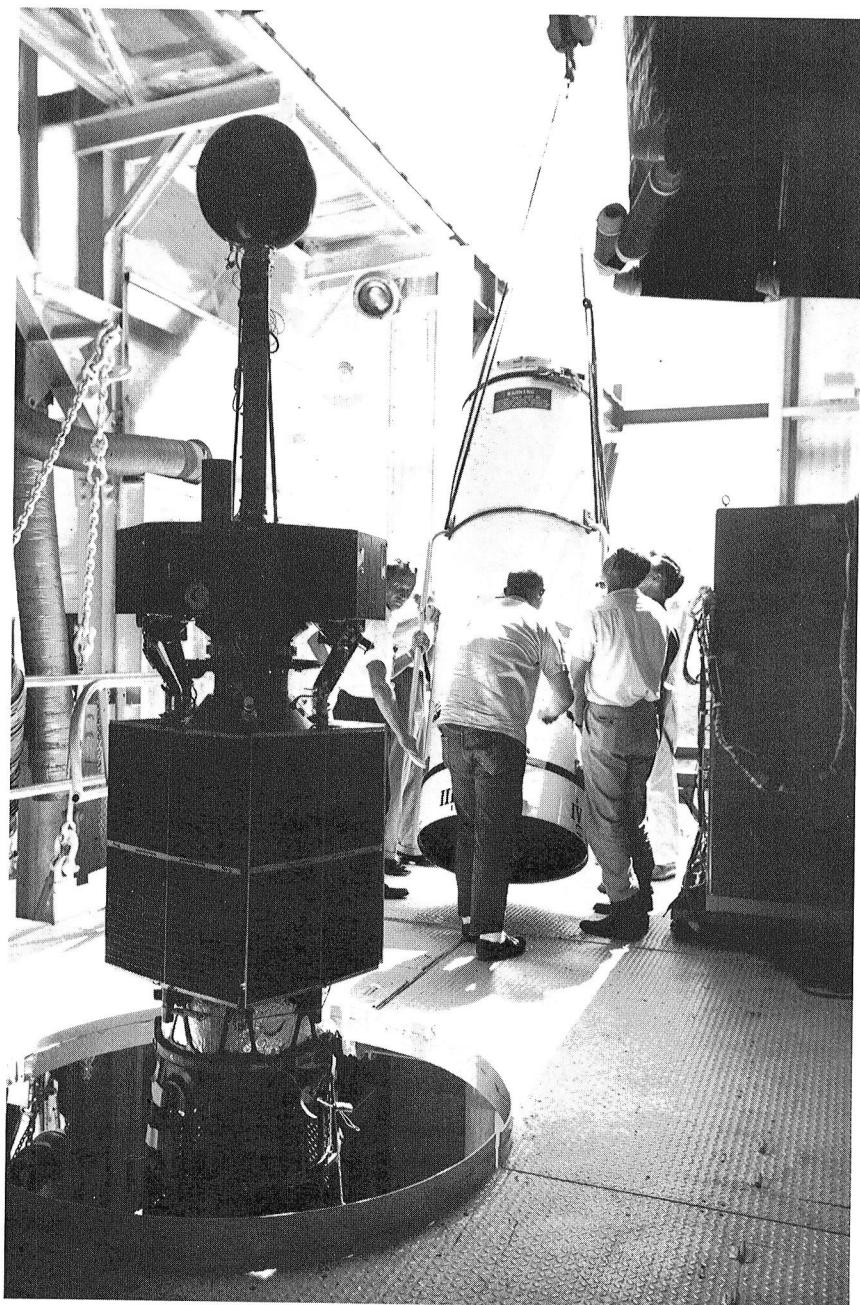


Figure 17. The Nose Cone (Fairing) for the IMP A Spacecraft Arrives at the Gantry to be Affixed to the Launch Vehicle.

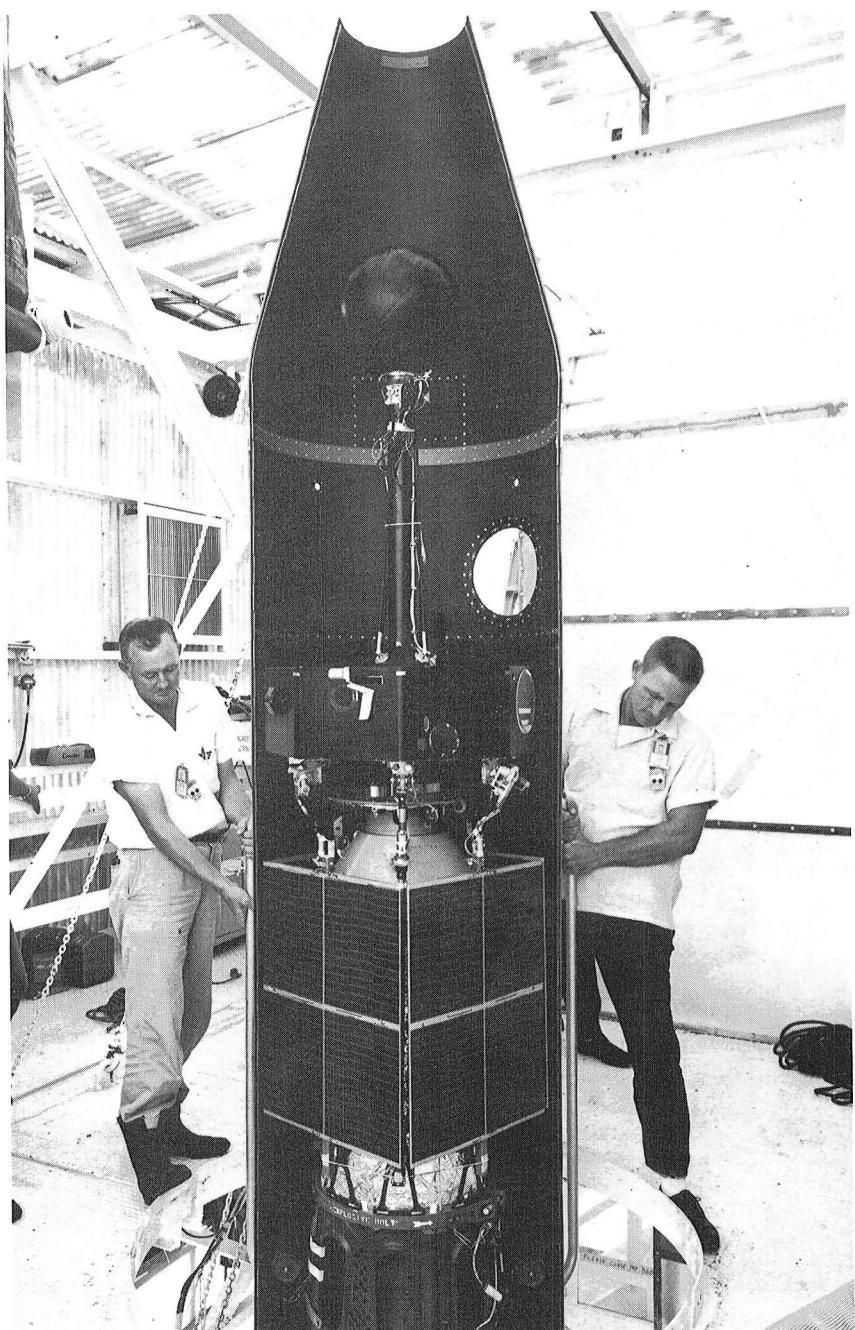


Figure 18. IMP A Nose Cone (Fairing) Installed on Launch Vehicle.

Chapter 3

The IMP Spacecraft, Its Orbits and Performance

IMPs A, B, AND C

The objectives of IMPs A, B, and C were to obtain scientific data in various Earth orbits that would assist in accomplishing the overall IMP mission objectives. To make sure this was accomplished, three spacecraft rather than one were developed and launched because of the many uncertainties in the early days of space exploration. These uncertainties included unknowns concerning the likelihood of successful launch, duration of useful life, effects of operating in cold environment caused by the Earth's shadow, and effects of operating in the vacuum of space.

Spacecraft Description

IMPs A, B, and C were almost identical spacecraft with each carrying the same experiments and virtually the same power supply and communications and data handling systems. Each was developed by the GSFC staff except for contract support on IMPs B and C from EMR Aerospace Sciences, Inc., College Park, Maryland. This support included mechanical assembly and electronic integration, and preintegration testing of experiments and instruments prior to their receipt by the electronics integration team. The contractor also monitored special testing of components, magnetic field tests, Sun-spin tests, calibration of experiments, and environmental testing of the system.

IMPs A, B, and C were each octagonal in shape and measured 71 centimeters (cm) across and 20 cm in height. Figure 19 illustrates IMP A. IMPs B and C were identical to this configuration.

The total weight of each spacecraft at time of launch was about 60 kilograms (kg), of which about 15 kg (on IMPs A and B) consisted of scientific instruments. On IMP C, about 20 kg of experiment weight was launched.

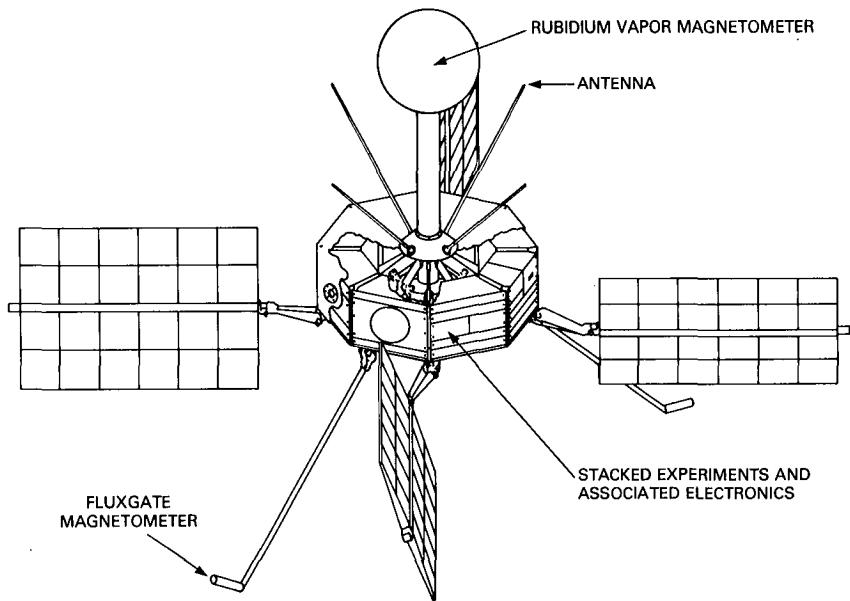


Figure 19. IMP A Satellite.

Most of the experiments and their associated electronic subsystems were stacked on a platform at the base of the structure which was fabricated from a fiberglass-aluminum honeycomb material (Figure 19). A magnesium tube in the center of the structure supported the platform and the total load of the spacecraft. It also provided the spacecraft interface with the launch vehicle.

Appendages

To minimize magnetic interference generated by the spacecraft, the fluxgate magnetometers on IMPs A, B, and C were mounted on the ends of two-meter (m) hinged booms which were extended after launch (Figure 19). The booms were mounted at the base of the structure, 180 degrees apart from each other.

The Rubidium (Rb) vapor magnetometer and its bias sphere were attached to a 0.9 m boom which, after fairing separation, could be extended an additional 69 cm from the top of the spacecraft along its

spin axis. The design of this boom posed special problems for the project mechanical engineer. The boom was designed originally as a fixed boom extending approximately 0.9 m from the main structure. The payload envelope inside the shroud determined the maximum length of the boom. Magnetic interference caused by the proximity of the spacecraft main structure and the Rb vapor magnetometer was encountered. To solve this problem, the boom was redesigned to allow the Rb vapor magnetometer and its bias sphere to be extended an additional 69 cm after the spacecraft was launched which effectively reduced the magnetic interference to acceptable levels.

De-spin System

A de-spin system was used on IMPs A, B, and C to reduce the spin rate of the spacecraft after rocket burn-out. The system consisted of two weights attached to a length of cable and hooked to the spacecraft 180 degrees apart and wrapped around the spacecraft opposite to its direction of rotation. Release of the weights was accomplished by firing redundant explosive dimple motors. The weights unwound in the same direction as rotation of the payload; the hooks were designed so that the cable released at the end of travel. This action imposed a torque on the payload resulting in a decrease in spin rate. The final spin rate was determined by the mass of the cables and weights and the length of the cables. Erection of the solar paddles and booms further reduced the spin rates to the desired orbital spin.

Power Supply System

IMPs A, B, and C each had four solar paddles. The paddles were 66 cm long and 46 cm wide, and were mounted 90 degrees apart from each other. On IMP C, the length of the paddles was increased from 66 to 70 cm. The IMP C spacecraft solar paddles were of a new design and the solar paddles were 2.4 kg lighter in weight. In addition, the solar cells were less susceptible to degradation from radiation.

On IMP A, each of the four solar paddles contained an array of 2,880 solar cells. These cells, when combined in a series-parallel network, supplied 38 watts of power at a potential of 12.0 to 19.6 volts and charged a 5-ampere-hour battery-pack consisting of thirteen 10-ampere-hour silver-cadmium cells. IMPs B and C used the same design. These batteries were selected because they were "magnetically" clean.

Battery Charge Control Circuitry

The batteries on IMPs A and B were subject to damage by overcharging. Because of a highly eccentric orbit, the spacecraft was in the Earth's shadow for only a small portion of each orbital cycle. Since the battery was needed for only a short duration on each orbit, it was subjected to long periods of charge. This excessive charging caused a build up of internal gas pressure in the cells forcing the electrolyte out of the cells resulting in battery failure. To alleviate this problem, a system was developed by GSFC engineers which used a two-level charge control for the IMP C spacecraft. When the battery required a charge current greater than 100 milliamperes (ma), an appropriate signal to the regulator set the voltage at 19.6 volts. When the battery charge current diminished to less than 50 ma, the regulator was set at 18.3 volts so as to eliminate the possibility of cell (battery) unbalance that would occur at the higher potential. The development of this technique was a valuable contribution to the following IMP spacecraft and to future spacecraft.

Under-Voltage System

On IMP C, the system voltage below which everything was turned off was changed from 12 volts to 11.5 volts to allow better operation at low temperatures. The recycle time was shortened from 8 to 3 hours because under normal conditions the battery could attain 90 percent recharge after a 3-hour period.

Communications and Data Handling System

Antennas

On each of IMPs A, B, and C, four 66-cm antennas were located 90 degrees apart at an elevation angle of 45 degrees. Affixed to the top of the structure body, the antennas rested on the inside of the fairing and were designed to lock into place at the predetermined angle after fairing separation.

The antenna system was a canted turnstile. The radiation pattern was pseudo-isotropic so that signals could be received from any spacecraft aspect observed by a ground station.

Transmitters and Receivers

On IMPs A, B, and C, the VHF radio frequency (RF) system was a unified system which provided a means for transmitting telemetry information, receiving and decoding commands, and transponding range

and range rate signals originated by a ground station. This system was the first to use the Goddard Range and Range Rate (GRARR) network. A ranging system was necessary to determine accurately the parameters of highly elliptic orbits.

The transmitter provided an output power of 4 watts with a sensitivity of -115 dbm. In normal operation, the system transmitted telemetry information to network stations. Upon interrogation by a GRARR station, however, the system transmitted the received ranging signals back to the GRARR station.

Transmitter Filter

On IMP C, a filter was added between the transmitter and the antenna to improve the sensitivity of the range and range rate receiver. This was required because of the great distance from the Earth to the spacecraft while at its apogee.

Telemetry and Data Systems

A pulse frequency modulation (PFM) telemetry system was selected for IMPs A, B, and C because of its relatively low power requirements.

Analog and digital data were electronically commuted by an encoder into a series of time-multiplexed PFM bursts and blanks (Figure 20). The output of the encoder drove the transmitter modulator. The frequency of the pulses on each group carried the information.

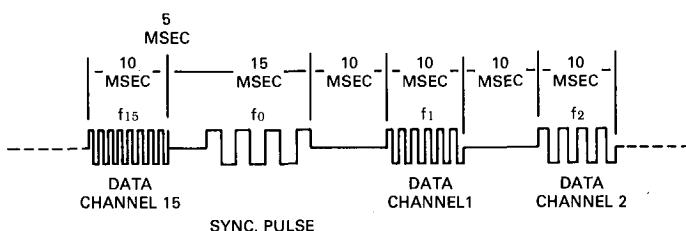


Figure 20. Pulse Frequency Modulation (PFM)

Performance Monitoring

IMPs A, B, and C carried on-board instrumentation designed by GSFC to telemeter 15 performance parameter (PP) measurements to ground control. These included four voltages, three currents and eight temperatures. In addition, two temperature measurements were "dual level", with the alternate level used to verify spacecraft separation from the third-stage motor and extension of the boom that supported the Rb

magnetometer and its bias sphere. Table 4 lists the measurements, locations and gives actual IMP A spacecraft operating data. Figures 21 and 22 illustrate the locations of PP and experiment sensors on IMP B.

Table 4
Performance Parameter Measurements

PP	MEASUREMENT	ACTUAL IMP-A SPACECRAFT OPERATING DATA
1	SOLAR ARRAY/BATTERY VOLTAGE	+19.6V
2	PRIME CONVERTER, +50V OUTPUT	+50V TO +51.5V
3	BATTERY CURRENT	<50 ma
4	SPACECRAFT CURRENT	1.8 AMP.
5	SKIN TEMP. NO 1 (TOP OF FACET D)	+15°C TO +30°C
6	Rb GAS CELL TEMP.	+50°C
7	BATTERY TEMP.	+15°C TO +45°C
8	PRIME CONVERTER, +12V OUTPUT	+12.2V \pm .2
9	SOLAR ARRAY CURRENT	2.7 TO 3.86 AMP.
10	SOLAR PADDLE (ARM #1) TEMP. (1)	0°C TO +10°C
11	SKIN TEMP. No. 2 (SIDE FACET 'D')	+10°C TO +20°C
12	MULTI CONVERTER, +7V OUTPUT	+7.2V
13	Rb LAMP TEMP. (2)	+100° TO +120°C
14	PRIME CONVERTER TEMP.	+25°C TO +45°C
15	TRANSMITTER TEMP.	+23°C TO +50°C

(1) ALSO INDICATED SPACECRAFT SEPARATION FROM X-258 THIRD STAGE MOTOR.
(2) ALSO INDICATED Rb MAGNETOMETER EXTENSION.

Thermal Control

Thermal control on IMPs A, B, and C was accomplished by using black and white paints, evaporated aluminum coatings, and highly polished surfaces.

On IMP B, some modifications were made in thermal design as a result of data obtained during the IMP A flight. The Rb vapor magnetometer bias sphere thermal control was changed to improve its flight performance. In addition, the passive thermal control system was adjusted so that the IMP B spacecraft would experience a slightly lower overall temperature in orbit.

On IMP C, the coating on part of the spacecraft was changed from a buffed aluminum finish to a white paint to reduce the temperature of the battery below that which was experienced on the earlier flight of IMP B.

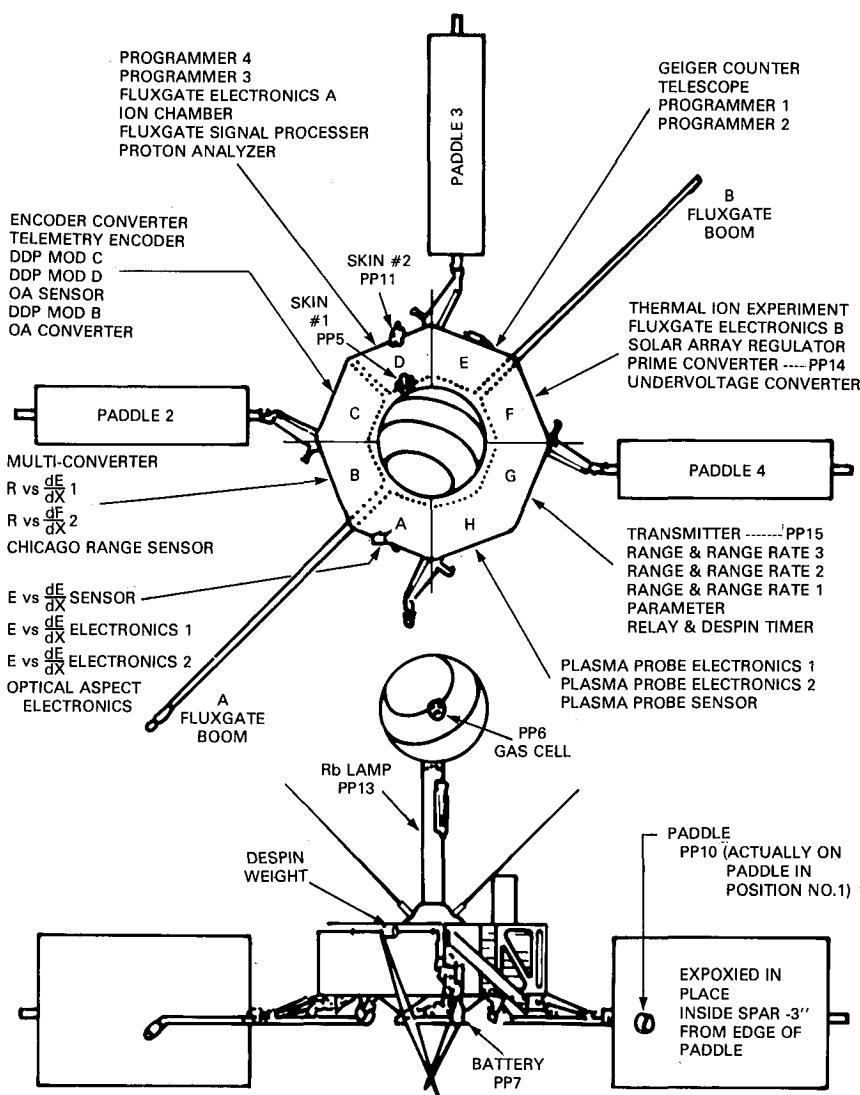


Figure 21. IMP B Temperature Measurement Locations (Top and Elevation Views).

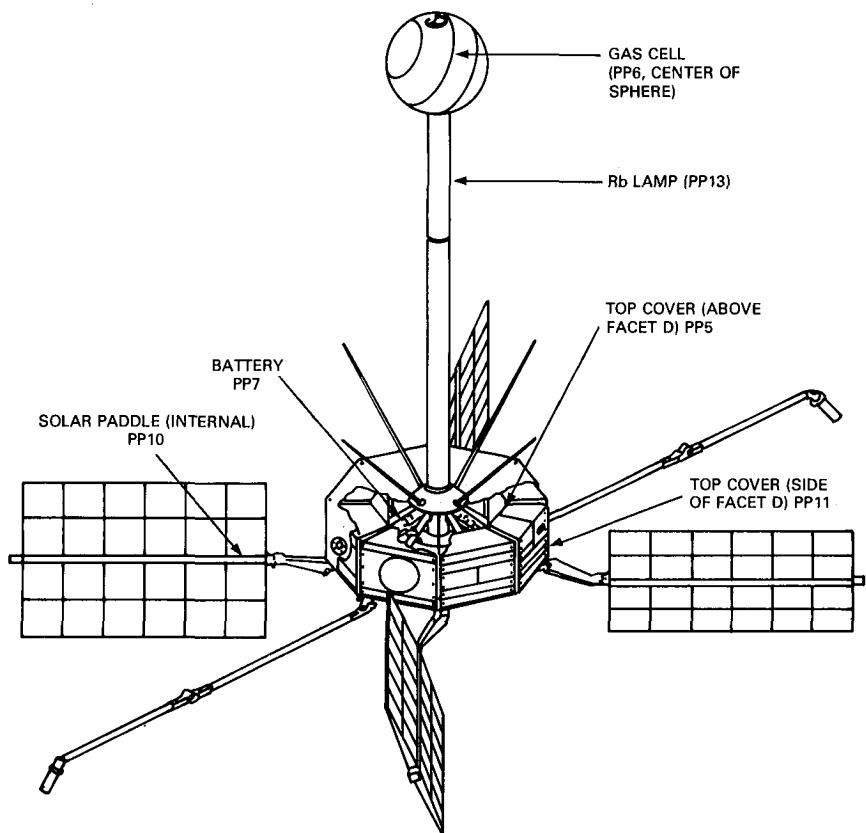


Figure 22. IMP B Temperature Measurement Locations (Perspective View).

Optical Aspect System

Each IMP satellite had an optical aspect system which incorporated two types of optical systems: a Sun-sensor system consisting of two digital solar aspect sensors, and a Moon-Earth sensing system consisting of three pencil-beam telescopes. The optical aspect telemetry data were processed on the ground to determine spacecraft orientation, spin rate, and shadow or sunlight condition.

Spacecraft Launch, Orbit and Performance

IMP A

IMP A was launched at 9:30 P.M. Eastern Standard Time on November 26, 1963, from the Eastern Test Range (formerly Atlantic Missile Range), Florida. The flight sequence was very nearly nominal and no problems developed with any of the spacecraft functions. Solar paddle and fluxgate boom erection, magnetometer extension, and separation of the spacecraft from the third stage motor occurred on schedule.

During and immediately following the launch, it was predicted that the apogee of the IMP A would be higher than planned. Early minitrack and range and range rate data indicated that the actual orbit achieved would be considerably lower than nominal, although within an acceptable limit. Sub-performance of the third stage rocket was determined to be the primary cause of the lower orbit. The lower apogee did not affect accomplishment of the scientific objectives of the mission.

Table 5 compares computed orbital elements with nominal or predicted values. Figure 23 illustrates the orbit achieved. Figure 24 depicts the IMP A orbit and the orbit-Earth-Sun relationship for 1 year after launch.

Table 5
IMP A Orbital Data

	Computed Data		
	11/27/63	1/16/64	Nominal
Apogee (km)	197,615.8	195,552.8	277,184
Perigee (km)	191.8	197.0	190
Period (min.)	5,666.0	5,583.2	9,164
Inclination (deg.)	33.338	33.336	33.003
Eccentricity	0.9376	0.9369	0.9547

Shortly after launch, the spin rate of IMP A was 22.27 rpm. Except for a slight decrease during the first orbit, the rate increased steadily, reaching 24.19 rpm 68 days after launch. The spin-up was attributed to solar radiation pressure acting on the solar paddles which produced a windmill effect. This spin-up continued for 151 days after launch, reaching a peak of approximately 27.5 rpm. In late April 1964, the spin

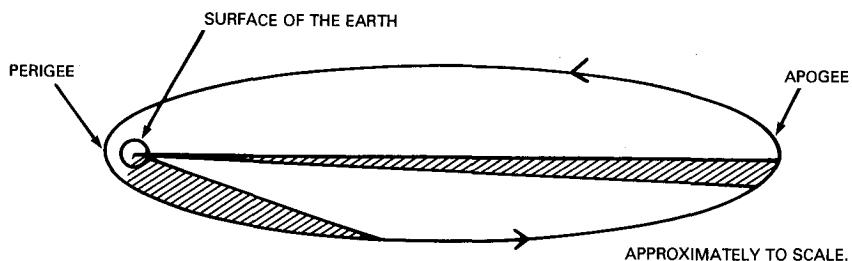


Figure 23. IMP A Orbit.

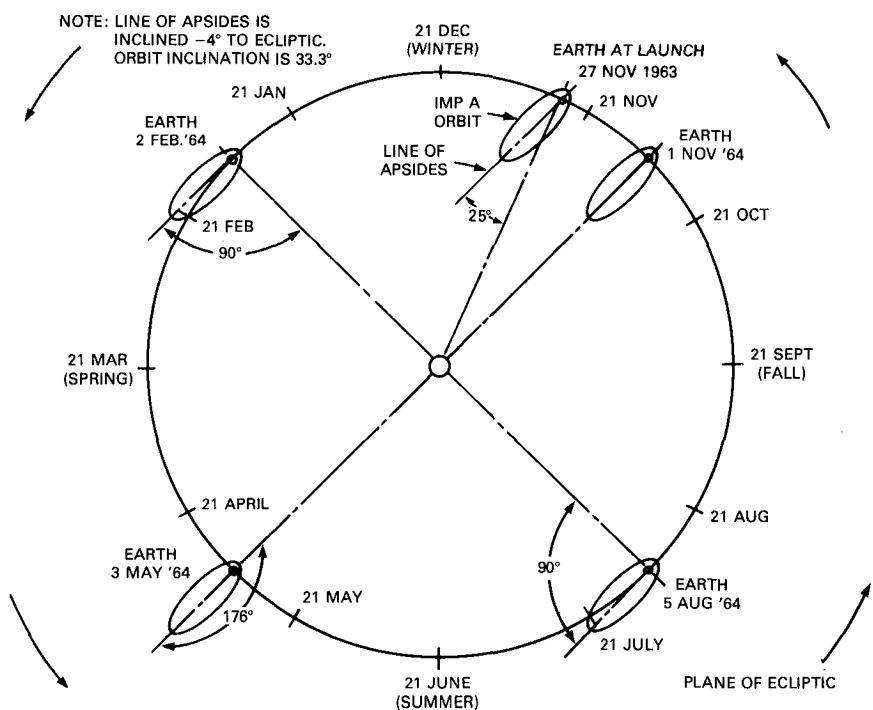


Figure 24. IMP A Orbit-Earth-Sun-Relationship.

rate began to decrease due to the Sun being above the spacecraft's equator. Spacecraft data indicated no detectable precession or abnormal motion of the spin axis.

During the period from launch through May 6, 1964, each of the experiments, with one exception, operated successfully and provided abundant and excellent scientific data.

The Thermal-Ion Electron Experiment functioned properly for the first 20 hours following launch, producing excellent data. Thereafter, the mechanical programmer, which changed the mode of operation, began to function erratically and only intermittent data were obtained for the following 3 days. The programmer then developed a repetitive pattern allowing the experimenter to recover approximately 10 percent of the data.

Beginning on February 3, 1964, a gradual degradation in one of two redundant circuits in a programmer card (gated telemetry amplifier) caused loss of data. This loss amounted to about one-half of the Rb vapor magnetometer data, but did not compromise the experiment's results.

Also in April, the Ames proton analyzer data became intermittent and the data indicated a slightly negative input to the encoder. These occasions occurred from 1 to 4 days apart at which time the data would be lost for periods from several hours to several days.

As predicted, on May 6, 1964, IMP A entered the Earth's shadow for an extended period of nearly 8 hours. This was a rarity that occurred only once a year. The temperatures on the IMP A outer surfaces fell to below zero as heat dissipated from the satellite. As a result of the extreme cold, one channel of the GSFC Cosmic Ray Experiment was lost. The failure was probably due to the photomultiplier tube, although a number of other items are possibilities. Future data from this experiment were of little value.

As the temperatures encountered by the satellite reached the maximum cold point of 60°C , the satellite shut off automatically and project officials were not sure if the spacecraft could survive. However, ground tests conducted at GSFC under simulated conditions indicated that the chances of survival were good.

IMP A emerged from the Earth's shadow and into the sunlight about 8 hours later and on May 7, 1964, a transmitter signal was picked up by a NASA tracking station in Santiago, Chile. IMP A had survived and was again operating.

Several days later, the spacecraft again turned off. Data taken from the Johannesburg, South Africa, tracking station indicated turn-off was

not instantaneous. However, normal spacecraft operation resumed about 12 hours later with no indication or explanation as to the cause of the temporary shutdown.

At the end of May, the spacecraft began to turn itself off and on in a repetitive series. The duration of the on-times gradually decreased during June from about $\frac{3}{4}$ hours to 1 minute or less. On July 14, 1964, Woomera, Australia detected an IMP signal for 2 seconds. Thereafter, data acquisition efforts were substantially reduced and later temporarily abandoned. Based upon the estimate of power output versus angle with the Sun and the seasonal change of this angle, it was predicted that conditions would be favorable in mid-September and again in November to support continuous transmission. The cause of the power problem was attributed to the degradation of the spacecraft battery. Proper operation would have continued except that the spin axis Sun angle was such that the solar paddles could not sustain continuous operation without an occasional assist from the battery.

On September 17, 1964, NASA's Space Tracking and Data Acquisition Network (STADAN) began a search for the IMP A satellite, and the Mojave, California station acquired and recorded an apparently normal signal. However, the on-off pattern was evident with the duration of on-periods varying from 30 minutes to several hours. The status of the spacecraft and experiments was essentially unchanged from that in May, except that noise was causing problems with some of the data from the Massachusetts Institute of Technology Solar Wind Experiment and the University of Chicago Cosmic Ray Experiment.

In October 1964, the duration of the operational periods decreased until only 1 minute was recorded on October 15. Tracking and data acquisition efforts were suspended until mid-November when the spin axis-Sun angle was expected to be favorable once again.

On November 12, 1964, the Mojave Station acquired and recorded the IMP signal for nearly 6 hours. Thereafter, and until December 15, 1964, the satellite operated about 90 percent of the time providing over 600 hours of data. The status of the experiments was unchanged from the previous operational period, except that the University of Chicago Cosmic Ray Experiment was not operating properly and the data was of no value.

A final period of IMP A operation began on February 21, 1965 and ended on March 25, 1965 during which time it provided intermittent and variable periods of operation. Small quantities of data were obtained during this time.

During the life of the IMP A spacecraft, a total of nearly 6,000 hours of data were obtained. The data indicated that the first spacecraft in the IMP series was highly successful and completed its portion of the overall IMP mission.

IMP B

The IMP B spacecraft was launched from Cape Kennedy, Florida on October 3, 1964 at 10:45 P.M. Eastern Standard Time. During launch, the first stage performance was above nominal. Second stage engine performance was satisfactory despite a slightly lower than nominal thrust level. A malfunction of the third stage occurred soon after ignition and resulted in reduced thrust and considerable coning of the third stage spacecraft configuration.

Because of the third stage malfunction, the injection velocity of the spacecraft was below nominal resulting in an apogee of 95,000 km (200,000 km was nominal). In addition, the dynamic perturbations introduced by the malfunction caused a shift of the spacecraft spin axis which resulted in a wider incident Sun angle. This situation resulted in low power output from the solar paddles and overheating of the spacecraft battery.

Thus, the low apogee attained and the spacecraft orientation in space was a direct cause of battery failure. Symptoms of the battery failure began to appear 2 months after launch. At that time, the battery proved incapable of sustaining spacecraft operation within the shadow of the Earth. A few days later, the battery failed completely. Thereafter, the spacecraft operated only during periods of favorable incident Sun angles and each time the spacecraft entered the Earth's shadow, it turned off for the 8-hour recycle period.

Although IMP B was injected into a lower altitude apogee than was intended, it provided useful data from within the magnetosphere. The spacecraft operated satisfactorily for 5 of its first 6 months in space. The experiments operated properly and significant data were obtained. However, due to the low orbit, it was not possible to observe the solar wind and to perform the interplanetary measurements. Thus, the primary objectives of the mission were not achieved.

The low orbit and the in-orbit orientation of the spacecraft with respect to the Sun resulted in an unplanned environment which exceeded the spacecraft design limits. Engineering data obtained from IMP B were used to improve the design of future spacecraft.

IMP C

IMP C was launched from the Eastern Test Range (ETR), Cape Kennedy, Florida at 7:00 A.M. Eastern Standard Time, on May 29, 1965. With the launching of IMP C, a turn-around time of approximately 7 months occurred (from the launching of IMP B). In approximately 18 months (from the launching of IMP A), three IMP spacecraft had been launched—a remarkable feat in launching of scientific satellites.

Table 6 indicates the IMP C orbital elements computed after launch.

Table 6
IMP C Orbital Elements

	Computed Data	
	5/29/65	<i>Nominal</i>
Apogee (km)	260,799	222,377
Perigee (km)	205.2	189.2
Period (min.)	8398.6	5959
Inclination (deg.)	33.87	33.01
Eccentricity	0.952	0.940

For the first 19 orbits, the spin axis was very stable and no coning was detected. On September 15, 1965, a slight coning was detected. A maximum cone angle of approximately 3 degrees was reached on October 5, 1965. The cone angle then diminished to zero on January 12, 1966 after the 40th orbit and remained at zero thereafter. It was thought that the anomaly was associated with radiation pressure torques acting on the spinning spacecraft.

The IMP C spin rate changed from 23.7 rpm initially to a minimum of 18.25 rpm after 80 days to a maximum of 27.4 rpm by 270 days to 22.4 rpm after 1 year. The spin rate variation was considered to be caused by the effects of solar radiation pressure acting on the solar paddles. The spin rate decreased when the Sun was above the spacecraft equator and increased when it was below.

While in orbit, the IMP C spacecraft returned a large quantity of useful data although several problems developed during flight.

One problem involved the failure of the MIT plasma probe experiment during the launch phase. Although the scientific data remained

abnormal, calibration and internal temperature measurements data were valid. Failure of the experiment modulator was suspected.

Another problem encountered was the failure of one of the GSFC fluxgate magnetometers. A review of the data indicated that the magnetometer was working properly immediately following spin-up of the spacecraft, but was not working following third stage burn-out. The failure of this magnetometer, however, did not seriously detract from the scientific results of the experiment.

A third problem was the lack of data from the Ames proton analyzer. During the initial orbits, no solar plasma was observed due to an unfavorable Sun angle caused by the tip-off of the spacecraft at injection. Failure of the experiments' high voltage power supply was suspected as the reason for not obtaining data in later orbits when Sun angles were favorable.

Two anomalies occurred in the University of Chicago Cosmic Ray Experiment data immediately after launch. The first anomaly, an apparent doubling of the expected counting rates, was due to a change in the type of detectors used in the IMP C experiment. The second anomaly involved the pulse height analyzer and was not considered to affect the interpretation of the data from the experiment.

AIMPs D AND E

The second group of IMP spacecraft were designed to obtain data on the lunar environment which the first group could not obtain. Thus, the primary objectives of the Anchored IMPs (AIMPs) D and E mission were to:

- Place the spacecraft into either a captured or anchored lunar orbit or a geocentric orbit with an apogee near or beyond the lunar distance,
- Investigate and obtain scientific data on the characteristics of the interplanetary plasma and the interplanetary magnetic field at lunar distances,
- Obtain data on dust distributions around the Moon,
- Obtain information on the lunar ionosphere magnetic field, lunar gravitational field, and lunar radiation environment, and
- Study geophysical and interplanetary phenomena by correlating AIMPs data with data from other spacecraft (OGO, Pioneer, etc.).

Because of the difficulties in achieving a lunar orbit in the early days of the space program, two spacecraft were developed. AIMPs D failed to achieve lunar orbit and entered into an Earth orbit, whereas AIMPs E was

successfully placed in lunar orbit. However, both satellites transmitted useful scientific data and met the mission objectives.

Spacecraft Description

AIMPs D and E were virtually identical spacecraft and were essentially the same as IMPs A, B and C. However, because of the requirement to launch the spacecraft into a lunar orbit, the structure for AIMPs D and E had to be reinforced to withstand increased loads imparted by a retro-motor.

The overall design, fabrication, integration and testing were performed at GSFC. The Westinghouse Defense and Space Center, Aerospace Division, provided integration and test support.

As with IMPs A, B, and C, AIMPs D and E each consisted basically of a two-piece magnesium axial-thrust cylinder having the third stage mating flange on one end and the fourth-stage retro-motor mating flange at the other end. Eight aluminum radial struts were attached to this cylinder and terminated at brackets on the periphery of the octagonal aluminum honeycomb platform, where most of the experiments and their associated electronics were stacked (Figures 25 through 28).

Appendages

Appendages were essentially the same as those in IMPs A, B, and C. The GSFC fluxgate magnetometers were inside fiberglass containers mounted on the ends of booms. The Ames low energy proton analyzer sensor was encased in an aluminum cannister to avoid possible magnetic interference problems.

Power Supply Systems

The power supply systems on AIMPs D and E were virtually identical. The power was generated from four solar array paddles. The four paddles contained a total of 7680 silicon solar cells. An average power of approximately 66 watts was generated initially (49 watts at the end of 1 year) at 19.6 or 18.3 volts across a two-state controlled shunt regulator. The power generated was delivered to both a 10-ampere-hour battery pack and to a prime time converter which supplied the basic regulated power to the spacecraft system.

Communications and Data Handling Systems

The communications and data handling systems for AIMPs D and E were virtually identical and were similar to those used in IMPs A, B, and

C. The systems provided a means for transmitting telemetry information, receiving and decoding commands, and responding range and rate signals originating from a ground station. A redundant command system was used to provide a back up in the event of failure.

Antennas

The AIMP antennas consisted of two major parts with interconnecting cables. The first part consisted of a diplexer/coupler assembly. Signals from the telemetry transmitter and the backup command receiver

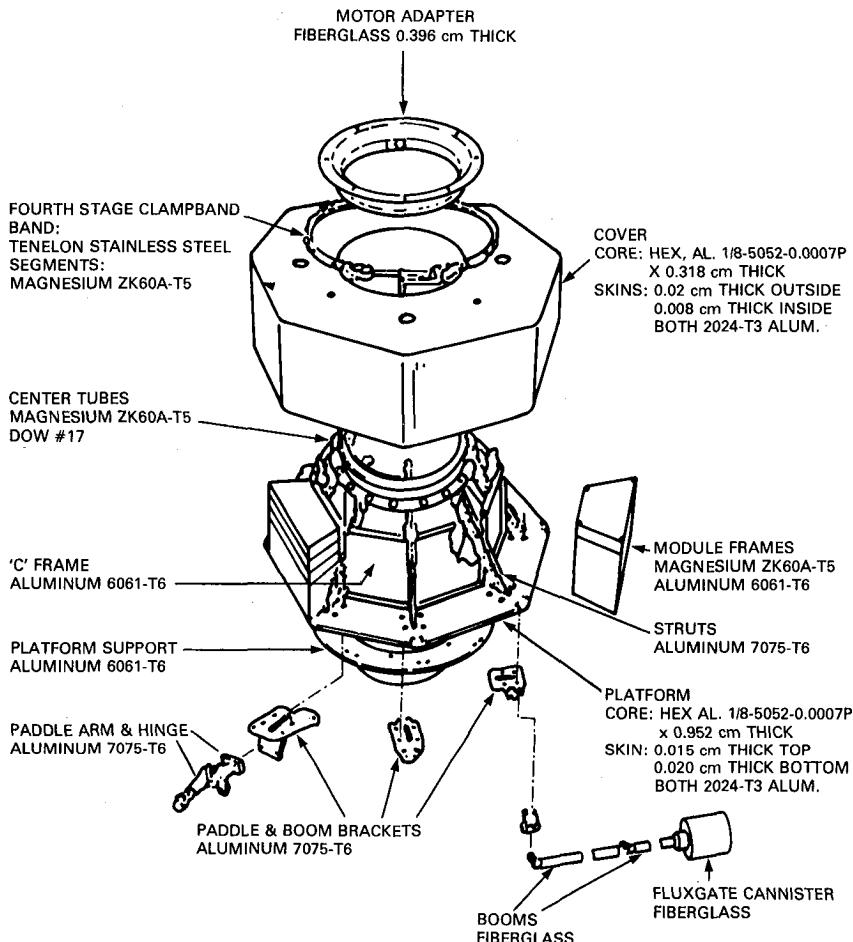


Figure 25. AIMP-D Structure (Isometric).

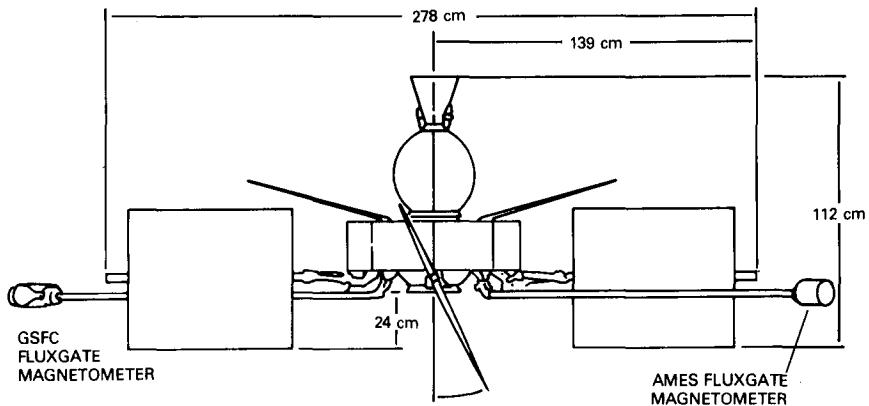


Figure 26. AIM-P Spacecraft, Side View.

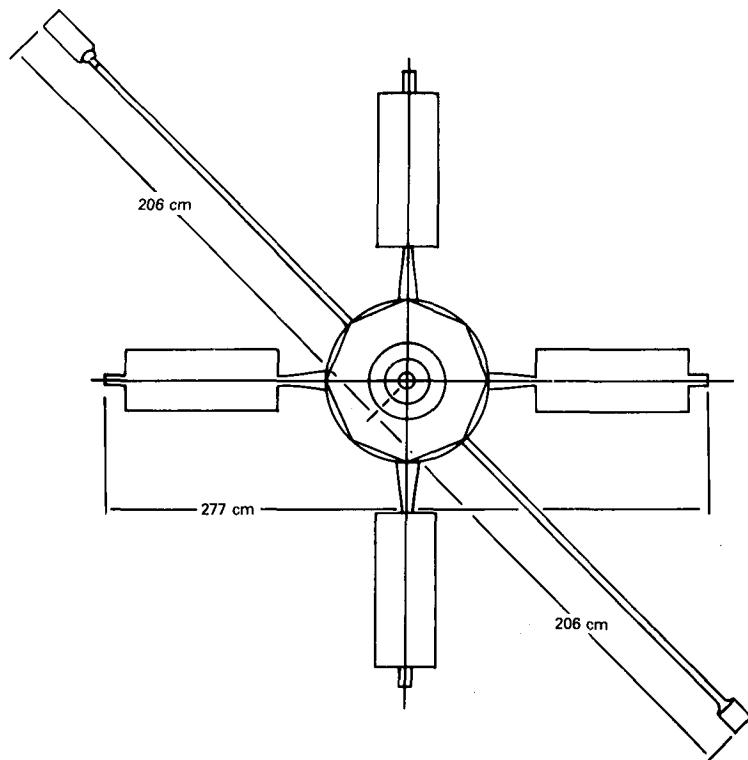


Figure 27. AIM-P Spacecraft, Top View.

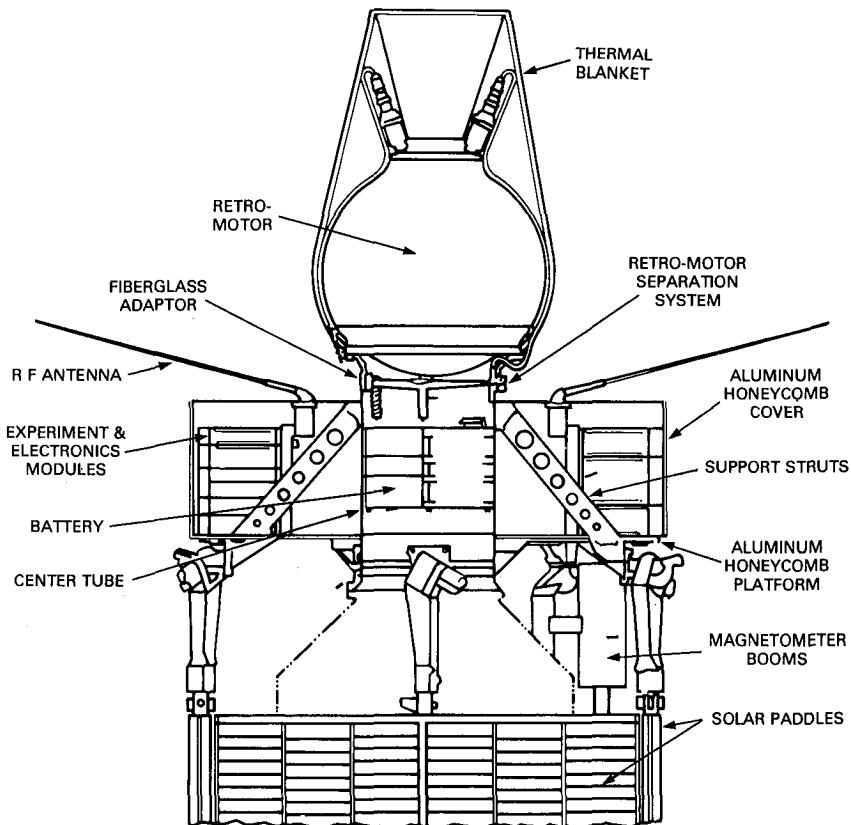


Figure 28. AIM-P Structural Cross Section In Launch Configuration.

were fed into the diplexer. The prime receiver was fed from the outer coupler port. The transmitter input was divided by the coupler into two equal signals phased 90 degrees apart which were fed to the antenna.

The second part consisted of four dipole antennas fed as a tilted-turnstile system. The antennas were mounted on the cover of the spacecraft at an angle of 15 degrees to the surface.

Command Subsystem

Commands were received by the spacecraft on a frequency of 148 MHz, which was phase-modulated by command codes. The detected command signal was fed into a command decoder which determined when any of the possible commands had been received.

Range and Range Rate

When it was desired to obtain the range or range rate of the AIMP D satellite, the ranging function was initiated by a 148 MHz signal. The ranging modulation was composed of a group of subcarriers which were 20, 4, 4.8, 4.16, 4.032 and 4.008 kHz. A modulation signal from the telemetry encoder was fed into the range and range rate system.

With AIMP E, the range data information transmitted from the spacecraft was centered around a sideband 807 kHz from the 136 MHz carrier. The range and range-rate transponder operated in conjunction with the spacecraft transmitter and one of the command receivers to provide ranging data.

Telemetry and Data Systems

In both AIMPs D and E, the systems included an encoder with a performance parameter measuring system. The telemetry transmitter had a power output of 7 watts and operated with a carrier frequency of 136 MHz. The carrier was phased-modulated by a square wave from the telemetry encoder. The transmitter signal was designed to be compatible with ground station phase-lock receiving systems.

Performance Monitoring

The performance parameter card in the spacecraft transformed essential currents, voltages, and temperatures into a voltage input to the encoder. Currents monitored varied from a small trickle charge of a few milliamperes to a maximum solar cell array current of 5 amperes. A self-saturating magnetic amplifier was adjustable according to the magnitude of the input signal. The output voltage from the magnetic amplifier was adjusted according to the telemetry, from 0 to +5 volts. The 12-volt main bus, battery voltage and the +28 volt transmitter were monitored.

Fourteen temperatures were monitored on the AIMP D satellite. Each sensor consisted of a thermistor-resistor network combination with a working range from -150°C to $+130^{\circ}\text{C}$.

Encoder

The encoder contained a sequence counter which is commonly called a "satellite clock". It consisted of a 16-bit accumulator. Once the sequence counter was referenced to a time standard, the time recorder by the telemetry station on the data tapes was compared to the sequency clock for time correlation. Two basic functions of the encoder were to:

(1) furnish the experiments and instruments with synchronized pulses to determine sample times and/or reference frequencies, and (2) convert the input voltages to a frequency that was used to modulate the RF carrier.

Thermal Control

Thermal control on AIMPs D and E was passive since there were no moving parts and no heat exchangers. Control was achieved through structural design considerations which allowed heat conduction paths such that temperature gradients and excursions were maintained within acceptable limits. To achieve thermal control, some surfaces were painted white and black while some were buffed aluminum. The retro-motor was encased in a thermal blanket.

The buffed, top-cover thermal surface on the AIMD deteriorated while in flight because of the exhaust from the retro-motor. Thus, a thermal insulation blanket made of Kapton and Tissueglas was designed for the AIMPE spacecraft.

Blankets had an advantage over painted coatings and buffed surfaces since their thermal control capabilities were less sensitive to the condition of the surfaces.

Optical Aspect System

The optical aspect system on AIMPs D and E was virtually identical to that used on IMPs A, B and C. The system had two modes of operation. In the transfer trajectory (from Earth to Moon) mode, three optical aspect readings were read out for every one Moon/Earth scan. The second mode occurred when a microswitch activated upon separation of the spacecraft and fourth stage (retro-motor). In this mode (Moon-orbit), seven Moon/Earth scan readouts for every one aspect readout were made.

Minor changes were made on AIMPE. A series of baffles were placed on the outer portion of the Sun sensor subsystem to eliminate any possible reflection of the Sun from the solar paddles.

Attitude Control System (AIMP E Only)

The attitude control system on AIMPE provided a means for reorienting the spacecraft prior to retro-motor ignition. The system also was capable of orienting the spin axis perpendicular to the plane of the ecliptic after the satellite was in its final orbit.

The system consisted of an ellipsoidal pressure vessel containing tetrafluoromethane (Freon 14) under pressure, and a high pressure line to a regulator that controlled the pressure to four solenoid valves. The resultant pressure was delivered to nozzles at the ends of the solar cell arrays. The nozzles were mounted on diametrically opposed solar array paddles with one nozzle pointing up, and the other pointing down. This arrangement produced a torque about the spacecraft's center of gravity.

Retro-Motor

The purpose of the retro-motor was to slow the spacecraft so that it would be acquired by the Moon's gravitational field and therefore injected into lunar orbit. The retro-motor, a Thiokol Chemical Corporation solid propellant rocket, was attached to the mating flange on the top of the spacecraft magnesium axial-thrust cylinder shown in Figure 28. The case material consisted of titanium, which is non-magnetic, to avoid possible magnetic field interference problems with spacecraft experiments.

Biological Decontamination

NASA's Office of Planetary Quarantine required that spacecraft with missions in the vicinity of the Moon be decontaminated. Existing facilities at GSFC were modified and a clean room complex was built to satisfy the requirements. The AIMP D and AIMP E proto-flight and flight units were assembled and decontaminated in Class 100 laminar cross-flow and down-flow clean rooms (see Figure 29). Decontamination was achieved using a system of alcohol baths, bacteriostatic conformal coatings, sterilization and heat cycles depending upon the subsystem.

Two methods were employed to recover viable organisms from the spacecraft surfaces to measure the effectiveness of the decontamination procedures. One method used exclusively on the module frames required a control strip attached to the frame. As each phase of decontamination was completed, a portion of the strip was removed and measured. The second method involved the use of sterile swabs and a template. The template was swabbed to obtain a sample from a known area, and a determination made of the number of viable micro-organisms present.

Due to the assembly of spacecraft units in controlled and clean-room areas, the AIMP E flight unit contained no more than 9×10^5 micro-organisms prior to decontamination, and no more than 2.7×10^4

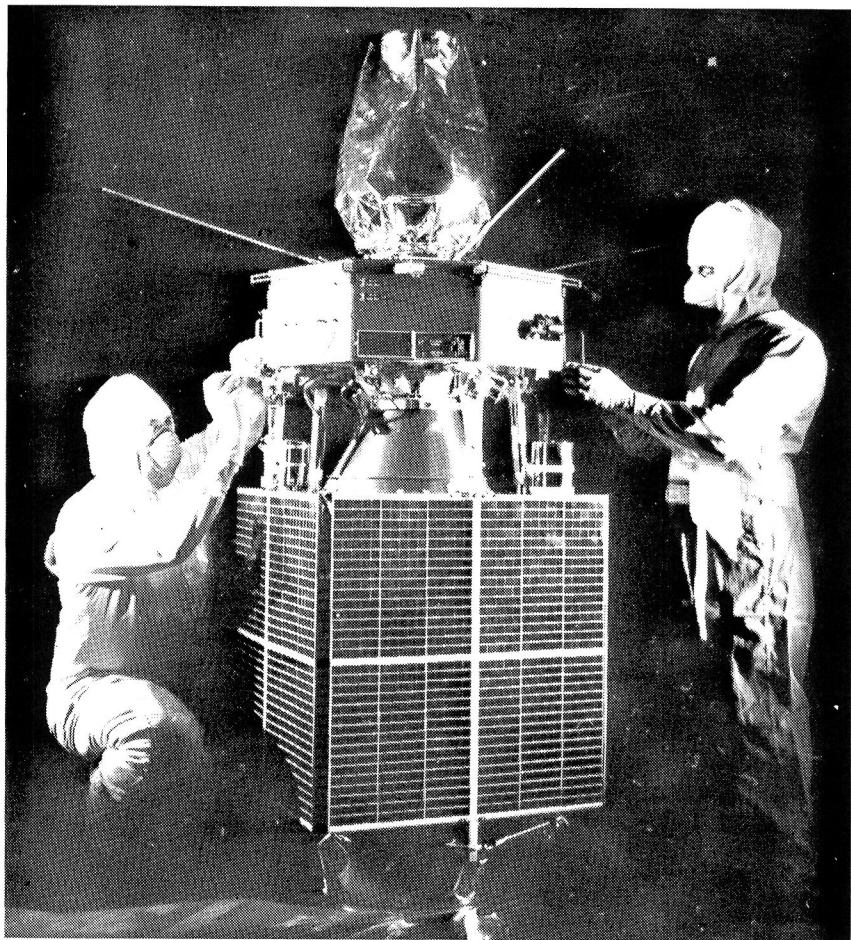


Figure 29. Clean Room Personnel in Protective Clothing Work on AIMPE Spacecraft.

micro-organisms after the decontamination process. This constituted a 97 percent reduction of organisms. An estimate of the number of viable organisms surviving internally in components was made based upon past history and known manufacturing environments. It was estimated that, of the total viable life remaining in the components, 10 percent would be spore forms. Of this 10 percent, approximately two-thirds would be aerobic and the remainder anaerobic. It was estimated that at the time of launch, the AIMPE spacecraft contained no more than 2.5×10^5 organisms. Of these, an estimated 2.2×10^5 organisms were contained inside of the components and foam encapsulant, and 2.7×10^4

organisms were on the surfaces. It was estimated that 7.4×10^3 of these organisms were spores.

Figure 30 contains a block diagram of the decontamination procedure at GSFC and the Kennedy Space Center launch site.

Spacecraft Launch, Orbit and Performance

AIMP D

AIMP D was launched from the Eastern Test Range, Cape Kennedy, Florida on July 1, 1966. The launch vehicle was an improved Delta three-stage rocket (Figure 31). The first stage booster was modified by the addition of three strap-on Thiokol solid propellant rocket motors. The booster was powered by a Rocketdyne liquid oxygen main engine. The engine was gimbal-mounted, to provide pitch and yaw control from liftoff to main-engine cutoff.

The second stage was an Aerojet-General Corporation liquid propellant engine employing inhibited red fuming nitric acid and unsymmetrical dimethylhydrazine as the propellants. The second-stage main engine also was gimbal-mounted to provide pitch and yaw control through second-stage burn. A nitrogen gas system using six fixed nozzles provided roll control during powered flight and coasting, and yaw control after second-stage engine cutoff. The second stage also had two fixed-nozzle deceleration and tumble motors to prevent a collision after second-stage separation.

The third stage was a spin-stabilized solid-propellant motor manufactured by the United Technology Corporation. The motor was secured on a spin table mounted to the second stage. The satellite was attached to the motor by means of a standard fitting which contained the third-stage sequencing and separation systems.

The spacecraft failed to attain lunar orbit because of a high-energy transfer trajectory. Unfortunately, all the small errors associated with the launch vehicle, though within prescribed limits, were in the same direction resulting in the high-energy transfer trajectory. Alternative missions were studied that would best satisfy the scientific objectives. It was decided that a highly eccentric Earth orbit having an initial apogee of 450,000 km and a perigee in excess of 30,000 km with a lifetime of at least 180 days would be acceptable for an alternative mission. The fourth stage (retro-motor) that was intended to place the satellite into a lunar orbit was fired by ground command after launch to achieve the alternative mission orbit.

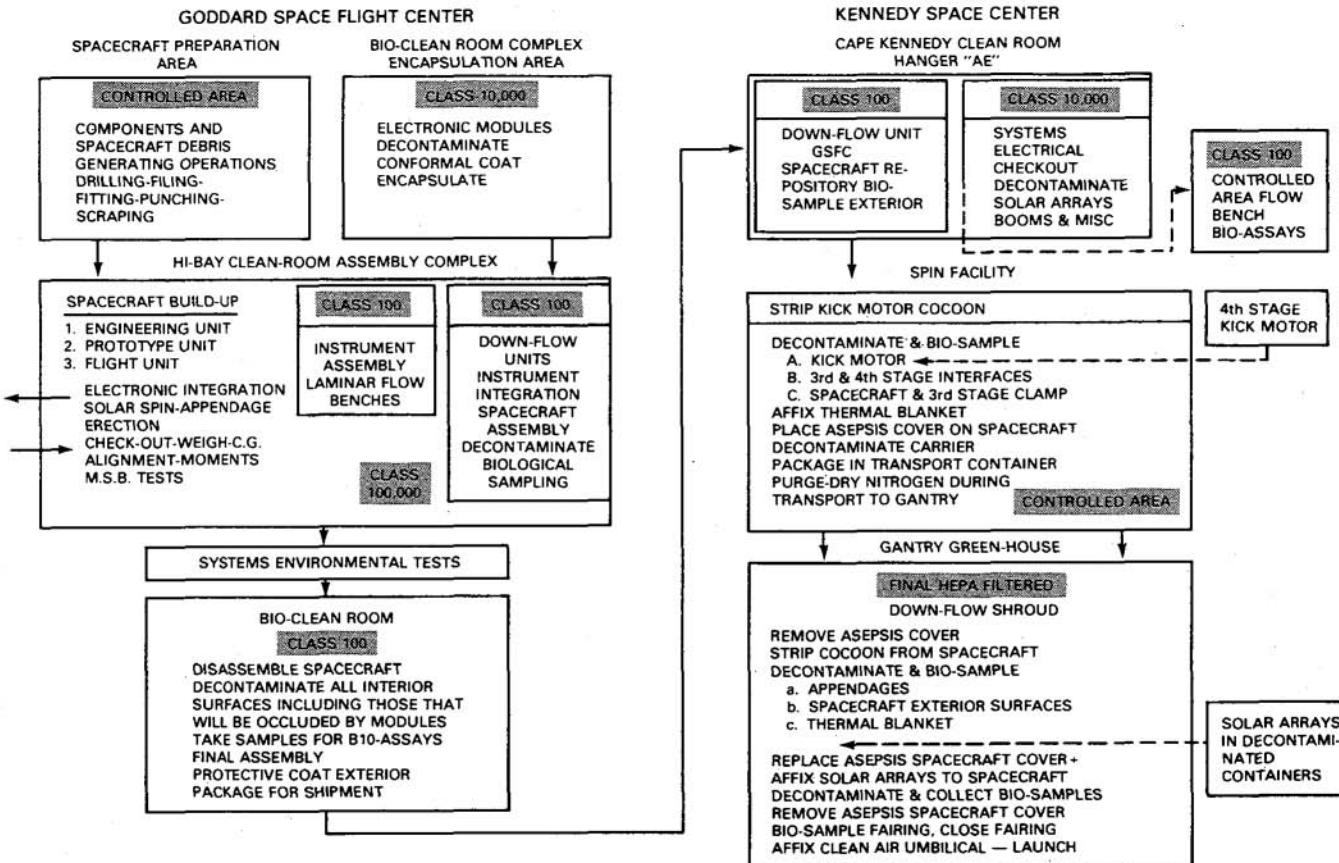


Figure 30. Decontamination Procedures.

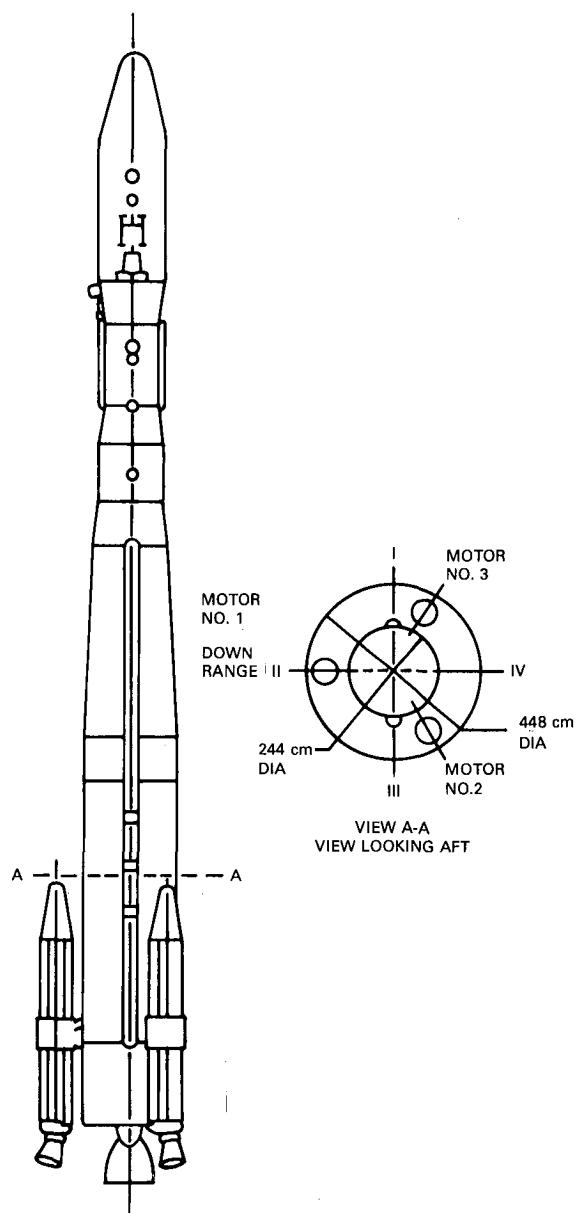


Figure 31. AIM-P D Launch Vehicle Profile.

The initial orbit achieved had an apogee of 450,000 km and perigee of 50,000 km, which was the highest elliptical orbit achieved at that time by an Earth-orbiting satellite. Table 7 lists the initial orbital elements.

Table 7
AIMP D Earth Orbit (Initial Values)

Apogee	450,000 km
Perigee	50,000 km
Inclination	28.5°
Period	15.5 days

During the following months, AIMP D made several close approaches to the Moon. Some of these are listed in Table 8.

Table 8
Selected Periods of Closeness to Moon

July 4, 1966	35,000 km
September 26, 1966	60,000 km
November 20, 1966	59,700 km
December 16, 1966	51,500 km
January 13, 1967	58,000 km
February 7, 1967	55,700 km
March 8, 1967	55,000 km

During the launch and trajectory phases, it was noted that there was a failure to the telemetered binary performance parameter bit. This bit (information) was supposed to indicate that the magnetometer booms were locked in orbit configuration. The problem was probably caused by failure of one of the micro-switches to function. When the retro-motor ignited, however, telemetry data indicated that the booms were locked and remained in this condition until retro-motor separation, when the telemetry data indicated that they were not locked. This indicated a marginal micro-switch closure condition (i.e., the switch may not have been entirely closed). The spin rate showed that the booms had been properly extended during the initial deployment.

An optical aspect anomaly was first noted after the spacecraft was in the orbit mode. Occasionally, in the telemetry data design for the spin period, a number equivalent to approximately 12 milliseconds ap-

peared. The 12 milliseconds corresponded to the length of a normal Sun pulse as was determined during spacecraft testing.

It was thought that a noise pulse on the trailing edge of the Sun pulse occasionally triggered the circuit. This appeared to be the next Sun pulse leading edge, thus giving an indication of measuring a Sun pulse width. The triggering circuit was sensitive to noise during the time of the trailing edge of the Sun pulse. The time interval between the appearances of this abnormal reading was several hours.

On August 23, 1966, the University of California Ion Chamber Experiment began to encounter periods of abnormal behavior. The two Geiger-Müller tubes showed all zeros on each readout. The Geiger-Müller tubes had failed and no further data were received.

Battery failure occurred on or about the 343rd day. The performance parameter which monitored battery charge current began to provide sporadic readings, which were initially hours apart. They became more frequent until the 348th day when the battery started to continually accept a charge.

The spin rate of the spacecraft was affected by apparent loss of mass of electrolyte from the battery. The spin rate was most affected on the 351st day and this was 2 days prior to the development of a major short in the battery. The battery temperature had its most dramatic rise from the 349th to the 354th day. Apparently whatever caused the temperature rise also caused the battery failure.

Turn-on anomalies appeared as the spacecraft entered the Earth's shadow for the first time on December 22, 1966 at 6:49 A.M. As the spacecraft exited at about 7:39 A.M. EST, it turned on for a few seconds. It remained off for nearly 4 hours and turned on again at 11:38 A.M.

The spacecraft entered the Earth's shadow for a second time on January 6, 1967 at 7:30 P.M. and exited at 8:11 P.M. EST. The spacecraft came on and remained on although there were voltage fluctuations in the power supply system. Tests were conducted at GSFC to determine the cause of the problem. By changing a capacitor value in the solar array regulator, the test system remained stable in all cases tested. These changes in the system were incorporated in the follow-on AIMPE spacecraft.

The University of Iowa reported that on the 203rd day, the quick-look telemetry data for its Electrons and Protons Experiment indicated that bits were stuck at 2^9 and 2^{12} . It was concluded that the digital data processor encoder accumulator for bit 2^9 failed. There was no reason to believe this failure affected other telemetry data. Two months later, another failure occurred.

During October 1968, after 27 months of successful operation, the GSFC Magnetometer Experiment ceased operation. It was believed that failure was caused by a power converter failure in the experiment or a protective fuse blew.

A power problem developed on the AIM P D spacecraft that would have caused a complete loss of spacecraft power had it not been resolved. The voltage dropped from the normal operating level of 18.2 volts to 13.5 volts. This caused some of the electronic systems to operate erratically. It was suspected that a short circuit had developed in the electrical wiring. After a review of the situation, the transmitter was turned off by a ground command in hopes that an increased power surge through the other electronic systems would "burn out" the suspected short circuit.

The spacecraft obeyed the turn off command. After a 40-minute waiting period (the time experts thought would be needed to overcome the suspected short circuit), a turn-on command was given. Telemetry transmissions were again received and the readings showed that the power level was back to the normal 18.2 volts. The spacecraft instruments and experiments operated as before the incident.

AIMP E

AIMP E was launched from the Eastern Test Range, Cape Kennedy, Florida, on a Delta vehicle on July 19, 1967, at 10:19 A.M. Eastern Daylight Time (EDT) (Figure 32). The launch vehicle for the AIM P E was similar in all respects to that used to launch the AIM P D.

The spacecraft attained a near-perfect lunar transfer trajectory. An investigation of possible lunar orbits was then conducted. It was determined to use the spacecraft cold-gas orientation system (attitude control system) to reorient the fourth stage (spacecraft and retro-motor) to obtain a lower periselene, and thus enhance the scientific data expected from the experiments.

On July 21, 1967, the first of three reorientation maneuvers began. The first maneuver was to move the spin-axis to increase the spin-axis Sun angle. This was done before retro-motor ignition in order to improve the lunar orbit characteristics. The Sun angle at the time of maneuver was known to be 120.75 degrees. The first pulse of the attitude control system put the spacecraft into a Sun angle of 122.75 degrees. A second pulse changed the spacecraft spin-axis Sun angle to 126.75 degrees.

The next day, the fourth stage (retro-motor) ignition was achieved, placing the spacecraft into the desired lunar orbit.

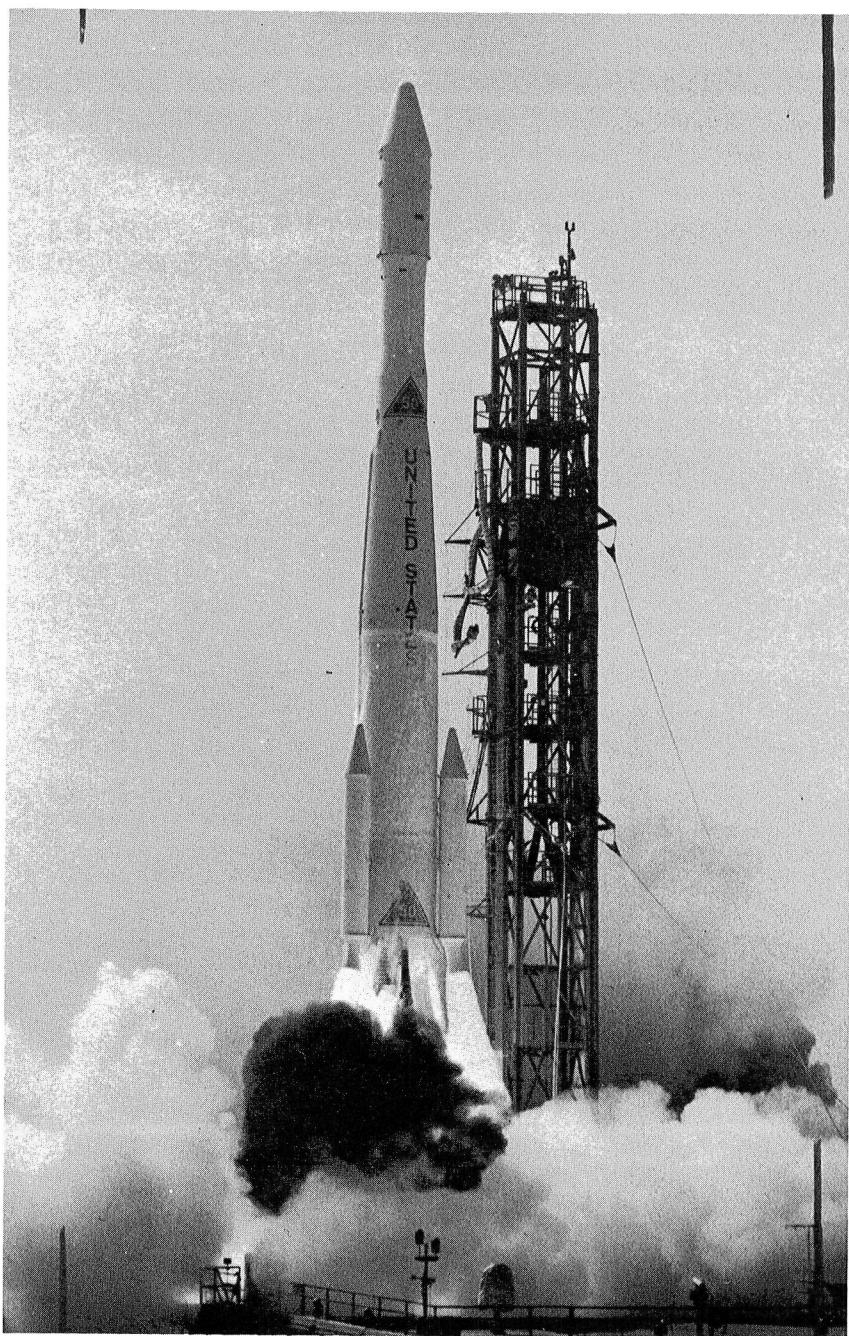


Figure 32. The AIMPE was Launched by the Delta Launch Vehicle July 19, 1967.

A few days later, the attitude control system was again commanded to reorient the spacecraft. This was accomplished in two steps because of the length of time required to verify commands. The first step was an in-plane maneuver to verify with the Sun-angle sensor that the reorientation system (attitude control system) was still functioning. The spin-axis Sun angle of the spacecraft had drifted back to 122.75 degrees since the last pulse commanded on July 21. After this first maneuver, it changed from 122.75 degrees to 120.25 degrees. The second step was an out-of-plane maneuver (defined as orthogonal to Sun and spacecraft spin-axis vector). A total of 36 out-of-plane commands were employed to complete the maneuver.

A day later, the third reorientation maneuver was commanded. This maneuver consisted of two additional out-of-plane commands and 16 in-plane commands. The final position of the spin-axis was calculated from data after completion of the maneuver. This position was within approximately 4 degrees of the south ecliptic pole. Thus, the attitude control experiment or reorientation system responded to each command transmitted and proved to be a highly acceptable and reliable system.

The injection velocity of the spacecraft into the transfer trajectory was slightly high, but within the acceptable limits of vehicle performance. This resulted in having a periselene higher than that desired for the scientific studies to be performed. A slight reorientation of the spacecraft spin-axis Sun angle prior to ignition of the retro-motor allowed the spacecraft to attain a lunar orbit that was very stable.

Table 9 shows the initial values of the lunar orbit.

*Table 9
Lunar Orbit (Initial Values)*

Aposelene	9429 km
Periselene	2538 km
Inclination	169°
Period	11.5 hours

The GSFC low energy integral spectrum measurement experiment (regarding potential analyzer) output started to degenerate 9 hours after the spacecraft was inserted into lunar orbit. The spacecraft performance parameters showed no variation before, during, or after this time. The temperature of the experiment was 30°C and good data were

obtained until the malfunction occurred. The malfunction did not result in a complete loss of experimental data; usable data recovery was between 15 and 20 percent. However, during the week of January 7, 1968, the experiment returned to full operation, and operated satisfactorily for the remainder of the year.

The University of California Energetic Particle Experiment ionization chamber portion during a solar particle event on November 20, 1968. However, the *Geiger-Müller* tubes operated satisfactorily.

IMPs F AND G

The objectives of IMPs F and G were to study solar and galactic cosmic radiation, the solar plasma, energetic particles within the magnetosphere and its boundary layer, and the interplanetary magnetic field. These two spacecraft were intended to continue and advance the studies initiated by earlier IMP spacecraft by making more detailed and precise measurements in Earth orbits. Whereas earlier IMP spacecraft were launched when solar activity (i.e., solar flares) was at a minimum, IMPs F and G were launched when maximum solar activity was expected.

This series of IMPs benefited from the advancements of previous IMPs, and as a result, the useful lifetimes were planned at 2 years each.

Spacecraft Description

Prior IMP spacecraft had accommodated more and different experiments with a minimum of redesign. However, with IMPs F and G, a new degree of sophistication in spacecraft structural and electrical design and in experiment complexity was achieved.

As with the earlier spacecraft, development of IMPs F and G was directed by GSFC. A support contract was placed with EMR Aerospace Sciences of College Park, Maryland to furnish three spacecraft, an Engineering Test Unit (ETU), a proto-flight spacecraft (IMP F), and a flight unit (IMP G). EMR also was responsible for the final integration of the spacecraft, testing, and final preparation for launch.

A non-flight spacecraft platform with harness was manufactured and integrated with spare IMP G components for use as a primary testing vehicle. This unit was designated the Component Test Assembly (CTA).

IMPs F and G consisted basically of a three-piece magnesium axial-thrust tube, an octagonal aluminum honeycomb equipment platform which was attached to the thrust tube and reinforced with eight radial struts, an aluminum honeycomb top cover, and four arms for supporting and orienting the solar cell arrays (solar paddles). The structure,

when assembled with top cover, was 71.1 cm across and 28.6 cm high. Figures 33 and 34 show the structure and experiment arrangement.

IMP G was identical to IMP F except for the following:

- Structural and hardware modifications were made to accommodate two new experiments and one new sensor;
- The IMP G magnetometer consisted of a single triaxial sensor on a boom with a dummy boom diametrically opposite to maintain a proper balance;
- Wiring changes were made to accommodate the new and/or modified equipment; and
- An RF shield was added to the top cover of the IMP G spacecraft to prevent the University of Chicago experiment, Range Versus Energy Loss, from picking up outside RF noise.

Appendages

Two hinged fiberglass magnetometer booms, each 1.83 m in length, were mounted 180 degrees apart from each other on brackets attached to the lower portion of the equipment platform. The magnetometer sensors were housed in fiberglass containers on the ends of the boom.

The loads on the magnetometer booms were increased by a new fluxgate flipping device employed for the IMP F mission. With this device, the X and Y sensors were mounted on one boom and the Z sensor with the flipping device was mounted on the other. Previously, on IMPs A, B, and C, each boom supported only one sensor. The diameter of the outer section of the booms was increased from 1.59 cm for IMPs A, B, and C to 1.91 cm for IMPs F and G. The inner section diameters were changed from 1.59 cm straight rod to a configuration which tapered from 2.54 cm at its base to 1.91 cm at its end.

Power Supply System

The power supply systems on IMPs F and G were identical. As in prior IMPs, power was generated from four solar array paddles. The paddles were attached to hinged brackets that were in turn attached to brackets 90 degrees apart on the bottom of the equipment platform. Each solar paddle measured 70.1 by 51.1 cm. Upon release from their folded launch configuration, the paddles first moved a short distance outward from the spacecraft, rotated into their intended orientation, and extended an additional 20 cm. When deployed, the solar paddles were oriented with respect to the spacecraft to produce maximum power when the spacecraft spin-axis was perpendicular to the ecliptic.

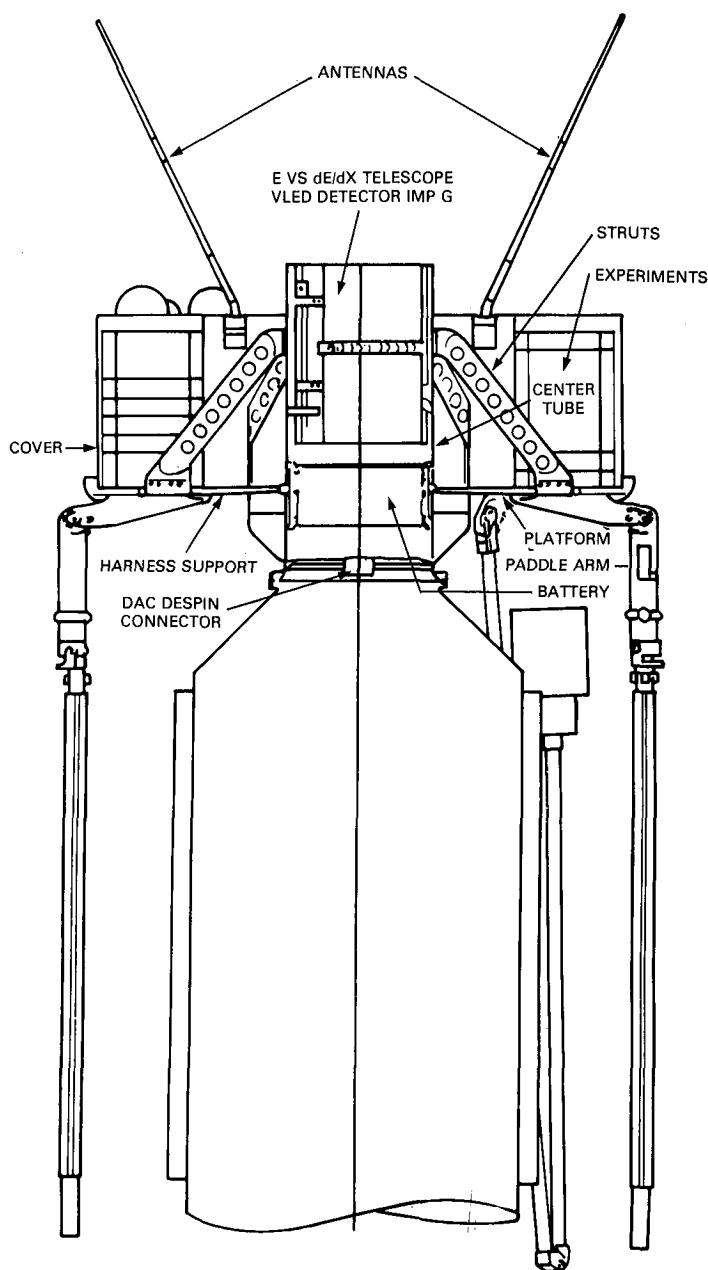


Figure 33. IMP F Structural Features.

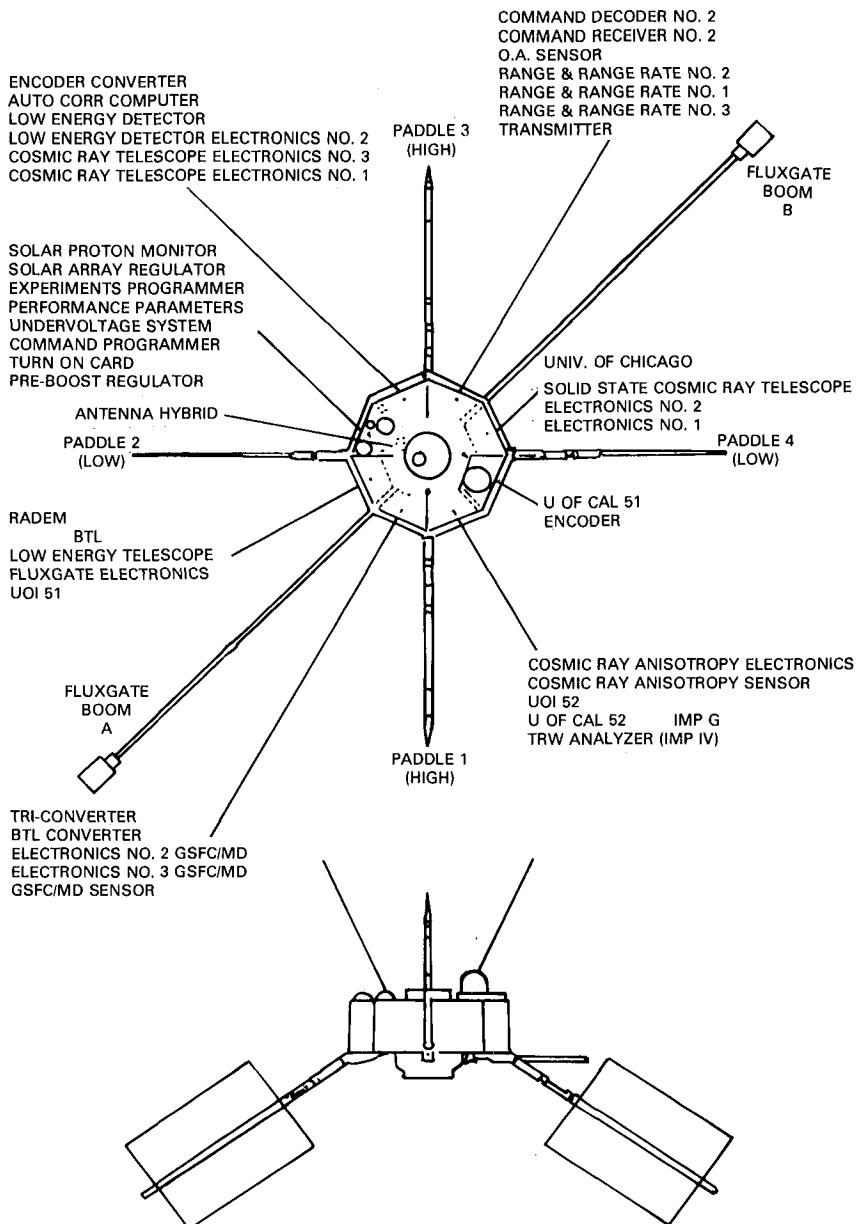


Figure 34. IMP F Placement of Experiments and Assemblies.

The solar-cell arrays were mounted on both sides of the paddles on an aluminum honeycomb substrate. The average weight of each solar paddle was approximately 2,454 grams. A total of 6,144, 2 by 2 cm, N-on-P solar cells were divided among the four solar paddles (768 cells per face). The cells were each covered by a protective cover glass to reduce radiation damage and provide thermal control.

A solar array voltage regulator was used to prevent excessive voltage generated by the solar cells from damaging the spacecraft experiments and battery. The regulator shunted excess current into an externally-mounted transistor-resistor "dump circuit" from which heat generated was radiated into space.

The storage battery was a sealed, non-magnetic silver-cadmium battery composed of 13 series-connected 5-ampere-hour Yardney cells. The nominal charge voltage for the battery was 19.6 volts with a reduction to 18.3 volts at the end of charge when battery current decayed to 75 milliamperes. An under-voltage detector switched the spacecraft current off when the battery voltage reached +12.0 volts.

Communications and Data Handling Systems

The communications and data handling systems used in IMPs F and G were almost identical, and were based on corresponding systems used on previous IMPs.

Antennas

Four dipole antennas, spaced 90 degrees apart, were located on the upper surface of the spacecraft top cover. These were set in cups at an angle of 25 degrees to the spin axis, and formed a canted turnstile.

The elements were fed in phase quadrature to produce a standard radiation pattern; i.e., left-hand circularly polarized from the top, linearly polarized from near the center, and right-hand circularly polarized from the bottom. The antennas were fed from a coaxial hybrid diplexer; the telemetry transmitter and one command receiver were connected to the antenna through the diplexer; the second command receiver was connected directly to the antenna. Transmitter output was fed through the hybrid to opposing antenna pairs by a half wavelength cable. The signals to each pair were equal in power but 90 degrees out of phase.

Range and Range Rate

When it was desired to obtain the range or range-rate of the spacecraft, the ranging function was initiated by a 148.260 MHz signal. The

range and range-rate were determined through the use of the Doppler principle as applied to the signal transmitted to and received from the satellite.

Telemetry and Data System

The pulse frequency modulation (PFM) telemetry data system was based on the PFM system used in IMPs A, B, and C, but was an improved design. It was a two-part system consisting of an encoder and a digital data processor (DDP) both housed in one package. The experiment capacity was approximately 460 bits, or four times that of IMP A. In addition, the digital bit rate was increased by a factor of ten over that employed in the IMP A using the same transmitter power at the same range. This was accomplished by increasing the number of bits per channel from three to eight and increasing the channel rate by four. Each burst was obtained from the output of a 16-level crystal-controlled frequency synthesizer with coherent frequencies.

The transmitter was essentially identical to that used in AIMPs D and E, except that it functioned at 136.140 MHz and 4 watts power output.

Performance Monitoring

The number of performance parameters (PP) measured on IMPs F and G was increased over the earlier IMP satellites because of a larger number of experiments and increased number of temperature measurements for evaluating the passive thermal control. Table 10 gives IMPs F and G performance parameters or measurements for the voltages, current and temperatures.

Thermal Control

Thermal control was achieved passively, through painted and evaporated aluminum coatings, as on prior IMP spacecraft.

Optical Aspect System

The optical aspect system used on IMPs F and G served three purposes: (1) to determine the orientation of the satellite in space as a function of time; (2) to provide sectoring of the satellite spin period for several experiments; and (3) to measure the spin-axis Sun angle and spin rate.

The system consisted of two basic parts: The satellite orientation system and the sectoring system. The orientation system was composed of an Earth sensor, a digital solar sensor, and necessary digital circuitry

Table 10
IMPs F and G Performance Parameters

PP	Measurement
1	TRW voltage (IMP F) U. of Iowa S2 level (IMP G)
2	U. of Iowa S1 level
3	Solar array current
4	Battery voltage
5	Battery charge current
6	Spacecraft current
7	Primary +11.7 volts
8	Primary +28 volts
9	BTL voltage monitor
10	MOSFET stability test voltage
11	RADEM
12	U. of Chicago voltage
13	Battery temperature
14	Pre-boost regulator temperature
15	Solar array temperature
16	Transmitter temperature
17	BTL temperature
18	U. of California temperature
19	U. of Chicago temperature
20	U. of Iowa S1 voltage
21	Flipper heater temperature
22	Center tube cover temperature
23	Lower cone facet H temperature
24	Spring seat temperature
25	Top cover facet H temperature (skin 1)
26	Platform facet H temperature (skin 2)
27	+7.0 voltage
28	Side cover facet H temperature (skin 3)

to determine the times when the two sensors were excited by their respective sources. The sectoring system consisted of an aspect clock that provided the logic decisions necessary to divide the spin period into 64 sectors. This basic division generated the outputs necessary to meet some of the experiment requirements.

Spacecraft Launch, Orbits and Performance

IMP F

IMP F was launched into a highly inclined orbit on May 24, 1967, at 0705 hrs or 7:05 A.M. PDT from the Western Test Range, California. The launch vehicle configuration (Figure 35) was the same as that used to launch AIMPs D and E.

At launch time, all but three experiments were turned on. These three, the Goddard Space Flight Center/University of Maryland Plasma Experiment, the University of Iowa Low Energy Proton and Electron Differential Energy Analyzer, and the TRW Systems Group Spherical Electrostatic Analyzer, were commanded on successfully on May 26 and 27 and on June 1, 1967, respectively. After the TRW experiment was turned on, the door covering the experiment at launch time was fired open.

Two anomalies in the spacecraft systems occurred early in the launch phase:

- After the door was fired open and the TRW experiment turned on, the experimental data showed good calibration and background count; however, the instrument malfunctioned and no useful scientific data were obtained. The experiment was finally turned off after 65 days.
- The optical aspect Earth sensor apparently sensed a reflection from a fluxgate canister on the end of one of the booms. This was a minor inconvenience, and did not affect any of the scientific data.

The orbit was selected so that the spacecraft spin axis would be perpendicular to the ecliptic plane. Table 11 shows the initial Earth orbit achieved by IMP F.

Table 11
IMP F Earth Orbit (Initial Values)

Apogee	214,382 km
Perigee	242.1 km
Inclination	67.12 degrees
Period	4 days, 7 hours, 51 minutes

The performances observed during the orbital phase are described below.

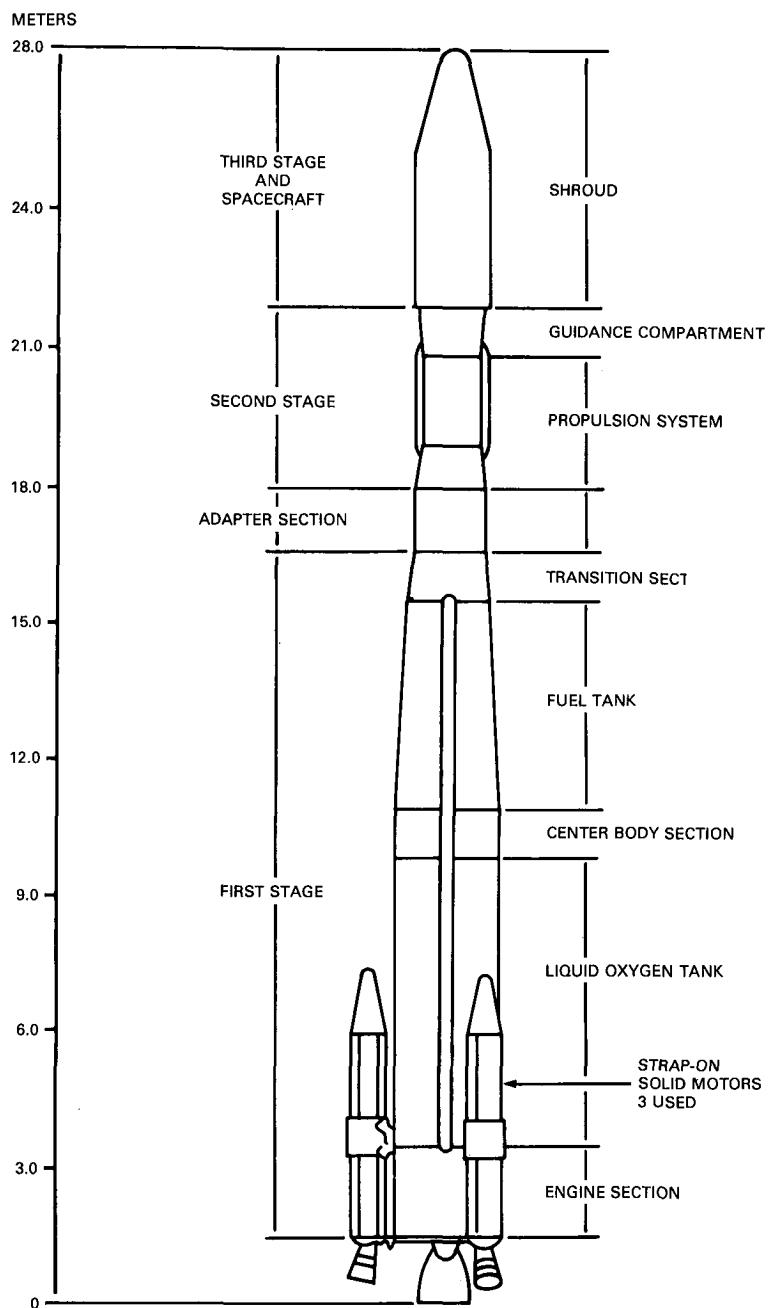


Figure 35. IMP F Launch Vehicle Profile.

The University of Chicago experiment, Range Versus Energy Loss, included an auxiliary telescope that had two data channels. After working for a period of several days, one channel deteriorated. This had a negligible effect on the experiment.

During the period from October 19-26, 1967, one Geiger-Muller instruments and systems functioned properly except for the TRW and University of Chicago experiments.

During the period from October 19-26, 1967, one Geiger-Muller (GM) sensor in the University of California Ion Chamber Experiment failed after three months of successful operation; however, one other GM sensor and the ion chamber operated satisfactorily.

In the first week of November 1967, the second GM sensor became intermittent and the ion chamber malfunctioned. As a result, the experiment was turned off. This was done on November 8, 1967. The experiment was turned back on again on November 13 but the data indicated that both GM sensors and the ion chamber were inoperative and the experiment was again commanded off. Several more unsuccessful attempts to obtain data were made in November and December 1967. In December, the experiment was commanded on and allowed to remain on. The data indicated that as of December 15, 1967, the ion chamber and one GM sensor were inoperative, and the other GM sensor was operating intermittently. The experiment was then commanded off and then on again after the completion of an orbit. Data taken from this maneuver indicated no significant change. The experiment was left on and in February 1968 the University of California reported that one of its two GM detectors and the ion chamber were providing sufficient data to warrant having the experiment remain on.

On March 7, 1968, IMP F entered an apogee shadow. Approximately 1 hour and 11 minutes after entering the shadow, the spacecraft turned off as a result of an under-voltage condition, and turned back on 8 hours later. At the time of turn-on, the readings from the optical aspect system were abnormal. Later data showed improvement, however, including periods of normal operation. In April 1968, the optical aspect problem cleared up and the system resumed normal operation.

GSFC personnel were concerned about the performance of the spacecraft and the experiments after being subjected to the prolonged freezing environment imposed by the March 7 shadow. Experimenters were requested to analyze the quick-look data received from the Fort Myers, Florida station from 0609 hours to 1207 hours GMT on March 10, 1968, and to report the status of their experiments. Minor problems were reported with two experiments:

- Failure of a photomultiplier tube in the GSFC Energy Versus Energy Loss Experiment resulted in the loss of the anti-coincidence count in the low-energy detector; useful data obtained from this experiment were reduced by 20 percent; and
- The University of Chicago Range Versus Energy Loss Experiment operated normally except for questionable sectoring.

The overall assessment of the passage through the apogee shadow was that by April 15, 1968, IMP F experiments and other systems were operating at an effectiveness level almost the same as that before entering the shadow.

IMP F passed through a near-apogee shadow between March 4 and 5, 1969. Post-shadow data indicated that the optical aspect system reverted to behaving in the same manner as it did during and after the March 1968 shadow; i.e., indications of constant Sun time, erratic spin period, and zeros for Earth time and Earth width. The experiment was left on, however, in hopes it would correct itself as it did after the previous shadow problem.

Telemetry data also indicated that degradation of the University of Chicago Range Versus Energy Loss Experiment increased by approximately 20 percent. Taking into account a 5 percent degradation caused by loss of the solar aspect function, the experiment operated at approximately 70 percent of its capability whereas it was operating at 95 percent effectiveness prior to passage through the shadow.

The spacecraft continued to transmit useful data until the evening of April 28, 1969, when it re-entered the Earth's atmosphere.

IMP G

IMP G was launched by a thrust-augmented Delta vehicle from the Western Test Range, California at 0147 hours PDT, June 21, 1969. The launch vehicle configuration was the same as that used to launch AIMPs D and E, and IMP F.

The orbit achieved (Table 12) was considered acceptable.

Table 12
IMP G Earth Orbit (Initial Values)

Apogee	176,826.8 km
Perigee	344.6 km
Inclination	86.8 degrees
Period	3 days, 9 hours, 6 minutes

The GSFC/University of Maryland Plasma Experiment did not function properly when it was first operated on June 23, 1969. The experiment was turned off and then turned back on for the second time on June 26, 1969. At this time, the background counting rate for the proportional counter in the Southwest Center for Advanced Studies Cosmic Ray Anisotropy Experiment increased. The Plasma Experiment was turned off on July 31, 1969 and remained off until August 18, 1969 when it was turned back on for 15 minutes for test purposes. The result was erratic operation of the plasma and several other experiments, and also the optical aspect system. The Plasma Experiment was considered inoperative and was turned off.

On August 10, 1969, after 43 days of good operation, the University of Iowa Low Energy Proton Differential Energy Analyzer suddenly ceased to operate. The stepping voltage and data readout went to zero.

On January 31, 1970, quick-look data indicated that the Bell Telephone Laboratories Low Energy Telescope Experiment was not operating properly. Several short turn-off, turn-on command exercises were performed in an attempt to clear the fault, but to no avail. The experiment was again commanded off and after two orbits was commanded back on; however, the fault remained. The experiment operated properly in several, but not all, modes. The cause of the problem was considered to be a failure in the experiment's power source regulator in the converter.

On December 20, 1972, the satellite experienced some drag. Three days later the signal from the spacecraft was lost as it disappeared below the horizon. IMP G re-entered the Earth's atmosphere at approximately 2300 hours. The spacecraft provided meaningful data during its 3.5 year life. At the time of its re-entry, nine of its twelve experiments were operational with only one of the nine degraded.

IMP I

The primary objective of IMP I was to investigate the nature of the interplanetary medium and the interplanetary-magnetospheric interaction during a period of decreasing solar activity. This included characteristics of the solar wind and interplanetary fields, and modulation effects on cosmic rays.

Additional objectives were to:

- Continue the detailed study of the radiation environment of cislunar space begun with the earliest IMP missions;
- Study the quiescent properties of the interplanetary magnetic field

and its dynamic relationship with particle fluxes from the Sun;

- Continue monitoring solar-flares;
- Extend knowledge of solar-terrestrial relationships;
- Further the development of relatively inexpensive spin-stabilized spacecraft for interplanetary investigations;
- Study the properties of low frequency radio waves from the terrestrial magnetosphere, the solar corona, and the Milky Way and determine their relationship to magneto-ionic properties of the solar system and the galaxy; and
- Monitor the electric field in the interplanetary medium.

Only one satellite was launched in this group. This was due to the greater confidence in achieving the desired orbit, greater confidence in the spacecraft reliability, and that a high performance and long life of the experiments could be expected due to past experience.

Spacecraft Description

IMP I accomplished a mission which represented a significant step forward in the evolution of the IMP Earth-orbiting spacecraft. The availability of a more powerful Delta launch vehicle allowed a larger and heavier payload to be carried. As a result, a sizeable increase in experiment weight, information rate, and available power was achieved.

There also was a change in the basic philosophy behind many of the experiments. Studies of magnetospheric processes were enhanced by using the same spacecraft for simultaneous measurements of radio frequency energy, energetic particles, and the Earth's magnetic field. IMP I was the first IMP that included RF energy studies along with fields and particles studies by various universities and government agencies.

The overall design, fabrication and integration responsibilities for the various in-house equipment systems were shared by the Spacecraft Technology Division and the Spacecraft Integration and Sounding Rocket Division at GSFC. Environmental testing of the IMP I was performed by the Test and Evaluation Division. EMR Aerospace Sciences of College Park, Maryland provided some integration support.

Geometrically, the structure of IMP I consisted of a 16-sided drum measuring 136 cm across the flats and 182 cm in overall height (Figure 36). As in prior IMP's, the structure of IMP I included an aluminum honeycomb shelf supported by eight radial struts. Figure 37 illustrates the structural configuration.

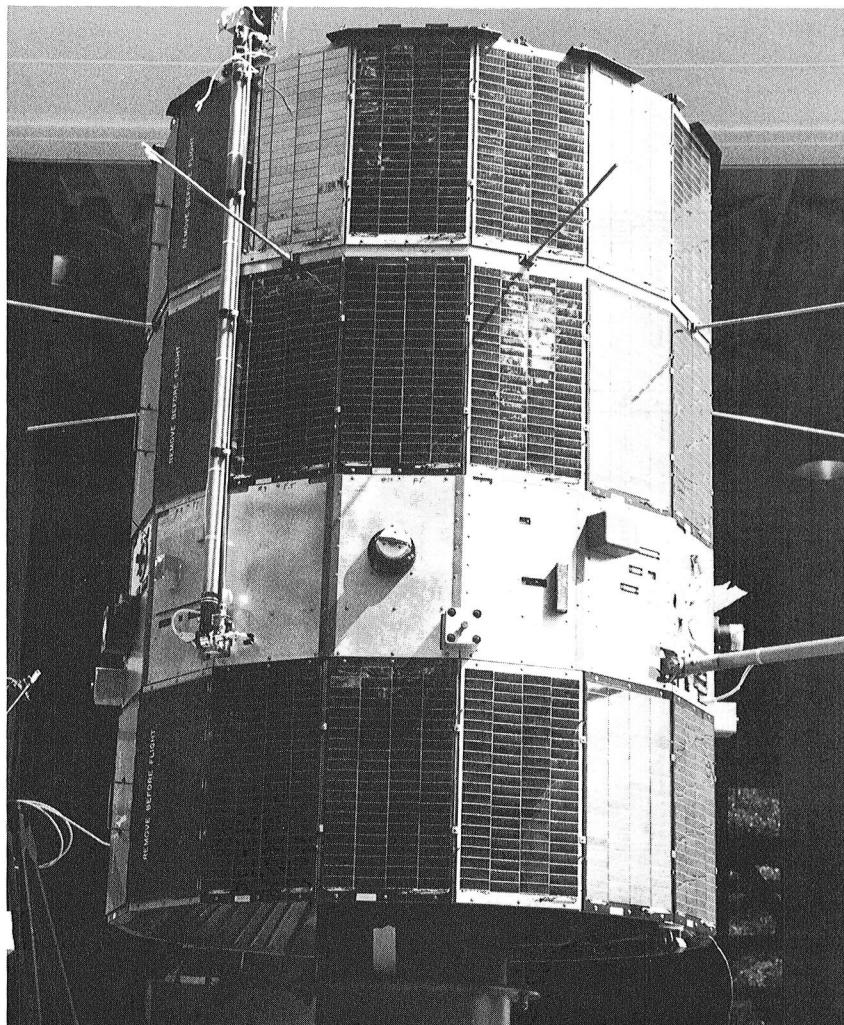


Figure 36. IMP I Spacecraft.

Appendages

Two attitude control system (ACS) booms each measuring 1.5 m in length were attached to the exterior of the spacecraft structure. These booms were designed to be folded alongside the spacecraft structure during launch and spacecraft separation from the third stage of the launch vehicle. They were deployed at a pre-selected time after separation.

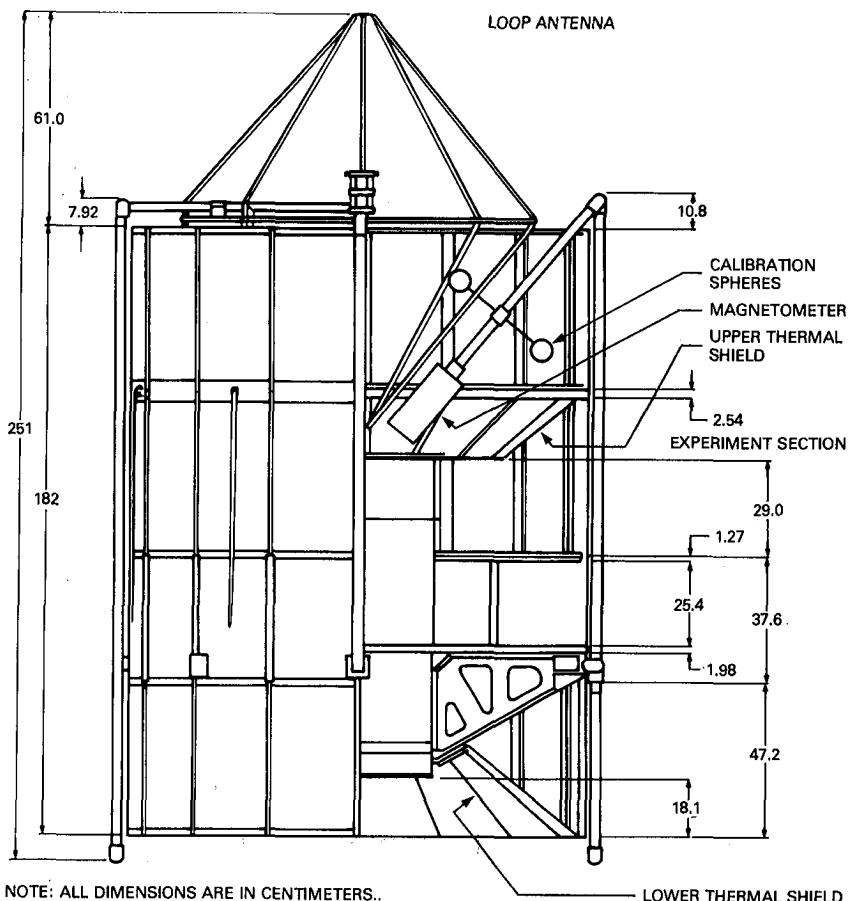


Figure 37. IMP I Structural Configuration.

Two experiment booms were attached to the exterior of the structure each 3.5 m in length. These booms were also folded alongside the spacecraft at launch and were deployed at a pre-selected time and sequence.

A loop antenna on a hinged boom was used which extended 448 cm from the spin axis of the spacecraft when deployed. Four electrostatic field measurement (EFM) antennas were provided; each was 61 m long when extended. Two other EFM antennas were also on-board; each extended from 3.1 to 6.1 m, with one deployed from the top and the other from the bottom of the spacecraft along the spin axis.

Figure 38 gives details of the spacecraft appendages.

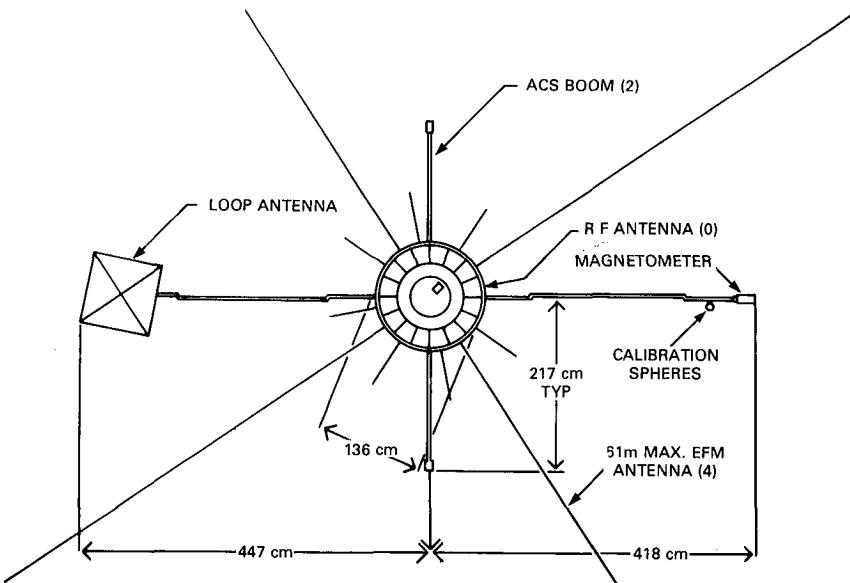


Figure 38. IMP I Appendages.

Power Supply System

As in prior IMPs, the power supply system consisted of body-mounted solar cell array panels, a silver-cadmium battery, and a solar array regulator. Forty-eight body-mounted solar cell arrays were used to convert solar energy into electrical energy. Energy conversion was accomplished by 4,032, 2 by 6 cm, N-on-P silicon solar cells. These cells were divided equally among the panels.

Minimum power generation of 128 watts at 28 volts was obtained under a combination of the following conditions:

- 1) When the spacecraft spin axis was perpendicular with the ecliptic within ± 10 degrees, and
- 2) Panel temperature was 17-18°C under solar illumination.

Solar Array Regulator

A shunt regulator prevented the solar array output voltage from exceeding 28.56 volts. Excess current was dissipated in the resistors and transistors which made up a "dump circuit".

Battery

A silver-cadmium battery was used on IMP I containing 14 series-connected 10 ampere-hours Yardney cells. The battery weighed 5.4 kg. Its maximum safe charge voltage limit was 21.1 volts. This limit was maintained by a battery charge regulator. The regulator was designed to reduce the charge voltage to 19.7 volts when the charge current dropped to less than 100 ma.

Communications and Data Handling System

The communications and data handling system design was based on those used in prior IMP's. The system was designed to transmit pulse-coded modulation (PCM) housekeeping telemetry data; analog, very-low frequency (VLF) University of Iowa experiment data; PCM computer data; and transponder Ground Range and Range Rate (GRARR) signals. It was designed to receive PCM/frequency shift keying (FSK) or tone-sequential commands and GRARR interrogation signals. A block diagram of the system is shown in Figure 39.

Antennas

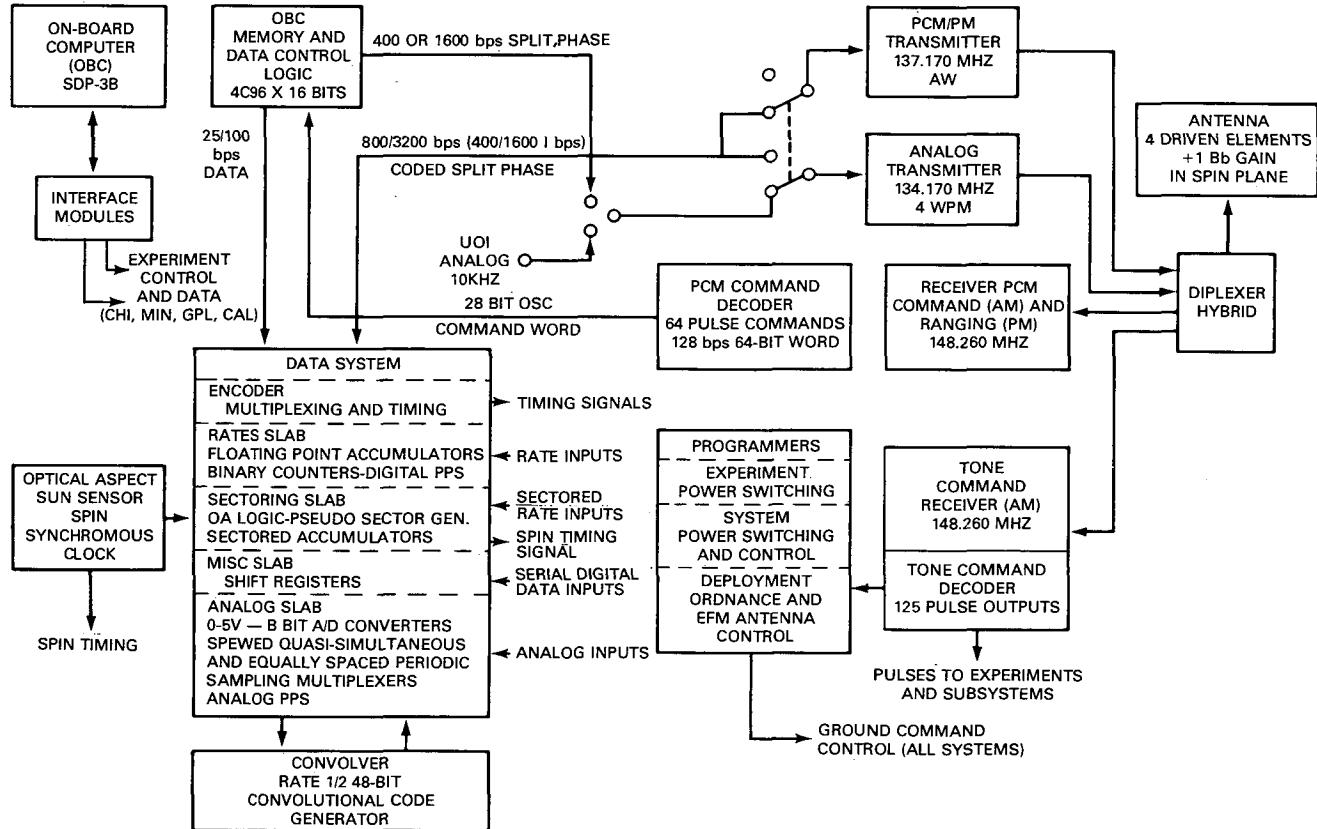
Eight telemetry antennas were mounted around the circumference of the spacecraft, 48.5 cm from the top of the spacecraft structure. The system consisted of alternate active and passive elements. The active elements were 53 cm long and were phased 0, 90, 180 and 270 degrees. The passive elements were 47 cm long and were mounted between the active elements. Each antenna element had a small chamber at its base which contained impedance-matching components. The antennas were fed from a coaxial hybrid system.

A diplexer system enabled transmitters and receivers to be used on the single antenna system. The system provided circular polarization along the spin axis and linear polarization near the spacecraft equator.

Transmitters

Two transmitters were used that were designed to operate simultaneously and provide PCM and analog experiment data transmissions to the Space Tracking and Data Acquisition Network (STADAN) stations.

One transmitter was used exclusively to transmit PCM convolutionally coded data. (Convolutional coding was a technique to transmit data redundantly in order to improve the signal-to-noise ratio.) It had a power output of 8 watts and a carrier frequency of 137.170 MHz. This frequency was modulated by the output of the spacecraft data system and had a peak phase deviation of 1.1 radians. This deviation caused the



transmitter power to be budgeted such that twenty percent was used for the carrier and eighty percent for the sidebands.

The second transmitter was an analog wide-band device. It had a power output of 4 watts and a carrier frequency of 136.170 MHz and was used in the different modes and submodes for transmission of range and range rate signals, University of Iowa analog data, on-board computer experiment data, and as a backup for transmission of the PCM telemetry data normally handled by the other transmitter.

Command Subsystem (Receivers)

IMP I was equipped with two 148.89 MHz receivers. One receiver received the encoded PCM and GRARR command signals and provided a ranging signal for the second transmitter, a detected pulse modulation (PM) signal for the ranging decoder, and a detected amplitude modulation (AM) signal for the PCM command decoder. The detected PCM commands were used to program the on-board computer experiment and to back up 62 of 125 tone-sequential commands.

The second receiver received the tone-sequential commands transmitted by the ground stations and the amplitude modulation (AM) signal for the tone-sequential command decoder.

Range and Range Rate

The first receiver and the second transmitter provided the exchange of range and range rate information between the spacecraft and a ground station. The ranging interrogation signals consisted of the assigned address tone followed by the ranging tones. Presence of the ranging signal was determined by the ranging decoder.

Telemetry and Data System

The IMP I telemetry system used pulse coded modulation (PCM). An encoding system was included which was designed to process, store, or multiplex data from each experiment until read out as split-phase PCM suitable for transmitter modulation. The telemetry and data system also included a digital and analog data processor.

The encoding system used approximately 200 logarithmic-compressing accumulators, 120 binary accumulators and 600 bits of shift register storage. The system was capable of storing 15,000 bits of data by using up to 400 metal oxide silicon field effects transistors (MOSFETS) per integrated circuit. Each accumulator included a binary counter with parallel transfer into buffer storage which could be shifted to produce a logarithmic compression. The result was stored until read out by telemetry.

The digital data processor performed several types of calculations. The spacecraft spin was divided into four, eight, or sixteen equal sectors, with an accumulator for each sector accumulating pulses for one or five spins. The total was then logarithmically compressed. The processor stored some events in the accumulators from the pulse height analyzer. When a higher priority event occurred before readout, the new data were accumulated and read out in place of lower priority data.

The analog data processor multiplexed 48 analog performance parameters and 13 experiment lines into one analog-to-digital converter with the results stored in 25 accumulators until read out.

Performance Parameters

More performance parameters were used on IMP I than on previous IMPs because there were more experiments in the payload. Sixteen analog performance parameters from a total of 48 were designed for standard thermistor sensors which were grounded externally to the encoder system. Two performance parameters were subcommutated with the encoder system for in-flight calibration and encoder parameter monitors. Thirty analog inputs with 110 k-ohm-to-ground input impedances were used by experiments and instruments. Table 13 is a list of analog performance parameters measured on the IMP I spacecraft.

Thermal Control

As in prior IMPs, the IMP I thermal design was completely passive, employing coatings on exposed external areas of the spacecraft structure in order to maintain the component temperatures within the design limits listed below:

- Solid-state detectors (except the University of California experiment) and battery: -10° to $+40^{\circ}\text{C}$.
- University of California detectors: one detector $\leq 20^{\circ}\text{C}$ and one detector $< 0^{\circ}\text{C}$.
- Spacecraft systems: -15° to $+50^{\circ}\text{C}$.
- Boom-mounted components: -40° to $+60^{\circ}\text{C}$.

Since the spacecraft had body-mounted solar cell panels, the only available external areas for thermal control were the mid-section panels and the upper and lower covers. The thermal coatings on these areas were as follows:

- Midsection panel: an external coating of black and white paint.
- Upper cover: black paint and buffed aluminum.
- Lower cover: an external coating of black paint.

Table 13
Analog Performance Parameters

APP Number	Function
1	Temperature, solar array
2	Temperature, battery
3	Distance deployed, antenna no. 1/guillotine no. 1 closure
4	Current, solar
5	Current, spacecraft
6	Voltage, spacecraft
7	Temperature, solar array
8	Temperature, low pressure gas
9	Current, battery
10	Voltage, battery
11	P.P., Haddock
12	Pressure, high (ACS)
13	Temperature, ACS tank no. 1
14	Temperature, ACS tank no. 2
15	Pressure, low (ACS)
16	P.P., Haddock
17	P.P., Bame
18	Distance deployed, antenna no. 2/guillotine no. 2 closure
19	Temperature, center tube (antenna system)
20	Temperature, platform (antenna system)
21	P.P./temp., Frank
22	P.P., Bostrom
23	P.P., Kellogg
24	Calibration encoder
25	Temperature, facet
26	Temperature, TM transmitter
27	P.P., Cline
28	P.P., McDonald
29	P.P., Simpson
30	P.P./temp., Anderson
31	Temperature, facet
32	Temperature, center tube
33	P.P. Anderson
34	P.P., Stone
35	P.P., Ogilvie
36	Distance deployed, antenna no. 3/guillotine no. 3 closure
37	Temperature, platform
38	Temperature, computer
39	Distance deployed, antenna no. 4/guillotine no. 4 closure
40	P.P., Aggson
41	P.P. Gurnett
42	Distance deployed, antenna no. 5
43	Temperature, Ness
44	Temperature, Gurnett (loop antenna)
45	Distance deployed, antenna no. 6
46	P.P., Bostrom
47	P.P., Frank
48	Cal/temp. encoder

All components and surfaces internal to the spacecraft were blackened to avoid large temperature gradients. The high-powered components were platform-mounted and located away from temperature-sensitive detectors. Provision was made to dump thermal energy from excess electrical power away from internal components. Black paint and buffed aluminum surfaces were used for boom-mounted components, including magnetometers and attitude-control assemblies.

The University of California experiment detectors operated at different temperatures. To achieve desired temperatures at the sensors, they were isolated from internal components and coupled to a separate, isolated section of mid-skin with a coating of vapor-deposited silver on Teflon.

Sun-Synchronous

The basic function of the spin synchronous clock (SSC) was to generate $2n$ pulses per spacecraft rotation period which divided the rotation period into $2n-1$ equal time intervals. The remaining interval was slightly less or greater than the others. The SSC used the pulse produced when the solar disk crossed the command slit of the Sun sensor to define the spacecraft rotation period and to reference the beginning of a new rotation period. The main outputs of the SSC included the pulse train containing the $2n$ pulses per rotation period, and the pulse defining the beginning of a new rotation period.

Optical Aspect System

Three parameters were measured by the optical aspect system: 1) the elevation angle with respect to the spin axis of two known references; 2) the time of the observation, and 3) the fraction of one spin period between each observation. Additional information, including spacecraft position with respect to two observed references, was available from Earth-based information. Figure 40 contains a block diagram of the optical aspect system.

Attitude Control System

The main purpose of the attitude control system (ACS) was to achieve proper spin-axis orientation and maintain the required spin rate. Modeled after the one used successfully on AIMPE, the ACS was modified to include both spin-up and de-spin capabilities as well as the spin axis orientation feature. Its first application during the mission was to align

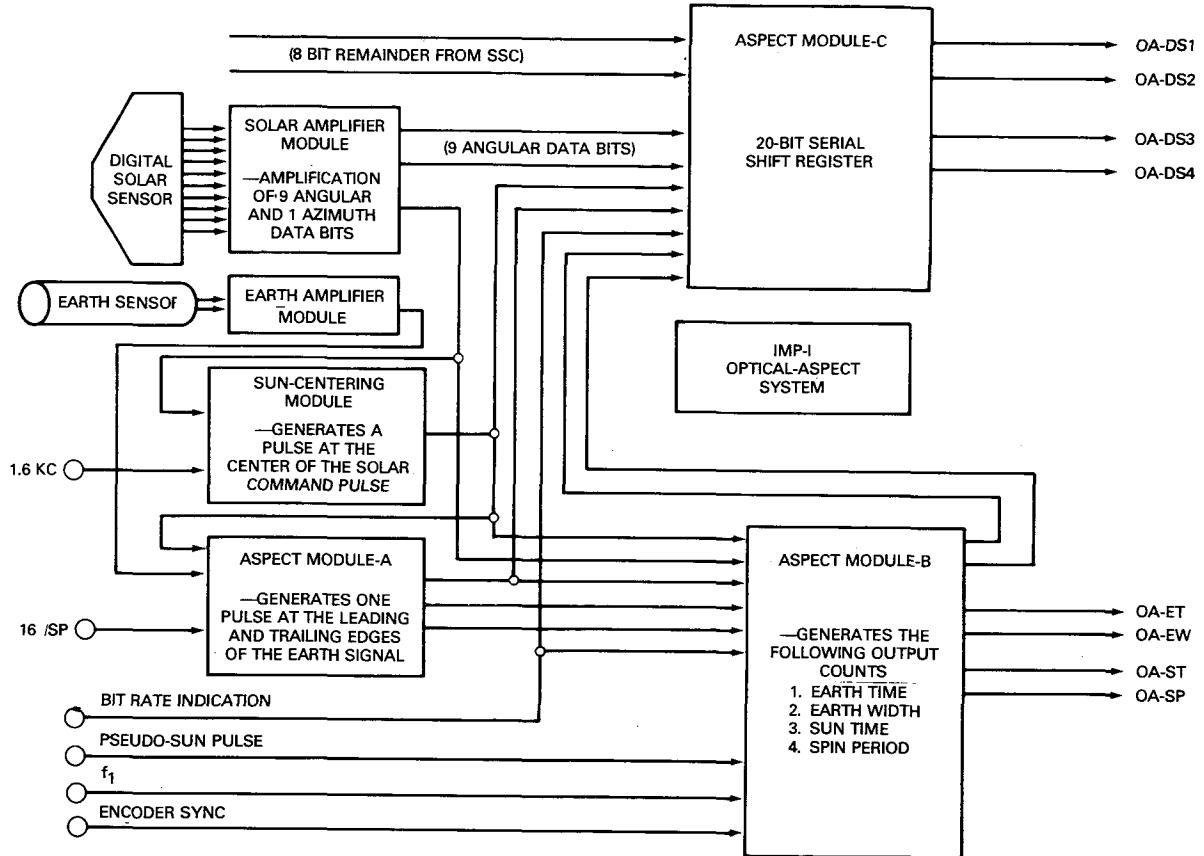


Figure 40. IMP I Optical Aspect System Block Diagram.

the spin axis perpendicular to the ecliptic plane. It was also used to maintain a 5-rpm spin rate during deployment of the electrostatic field measurement (EFM) antennas when the spacecraft moment of inertia increased by a factor of 35. The de-spin capability was required to maintain the spin rate if partial retraction of the EFM antennas was required.

Installed below the honeycomb shelf, the ACS consisted of two spherical tanks each containing approximately 3.6 kg of Freon-14 propellant and a single regulator to maintain a working pressure of 2.8 kg/cm². Solenoid valves permitted expulsion of propellant through expansion nozzles located on booms 2 meters from the spin axis.

Spacecraft Launch, Orbit and Performance

IMP I was launched on March 13, 1971, at 11:15 A.M. EST from the Eastern Test Range, Cape Kennedy, Florida on a Delta 3-stage launch vehicle (Figures 41 and 42). At 12:09 P.M. EST, the spacecraft was injected into a nearly nominal orbit.

The first stage was a McDonnell Douglas Corporation modified Thor booster with six strap-on Thiokol solid fuel rocket motors.

The second stage was powered by an Aerojet General Corporation liquid-propellant pressure-fed engine using inhibited red fuming nitric acid and unsymmetrical dimethylhydrazine.

The third stage was a spin-stabilized solid-propellant motor, manufactured by the Thiokol Chemical Corporation, which was secured on a spin table mounted to the second stage.

A standard Delta fairing was attached to the forward face of the second stage. The fairing protected the spacecraft from aerodynamic heating during the boost phase of flight and was jettisoned soon after the vehicle left the atmosphere.

Table 14 lists initial earth orbit parameters.

Table 14
IMP I Earth Orbit (Initial Values)

Apogee	206,258 km
Perigee	243 km
Inclination	28.69 degrees
Orbital Period	4 days, 4 hours, 32 minutes

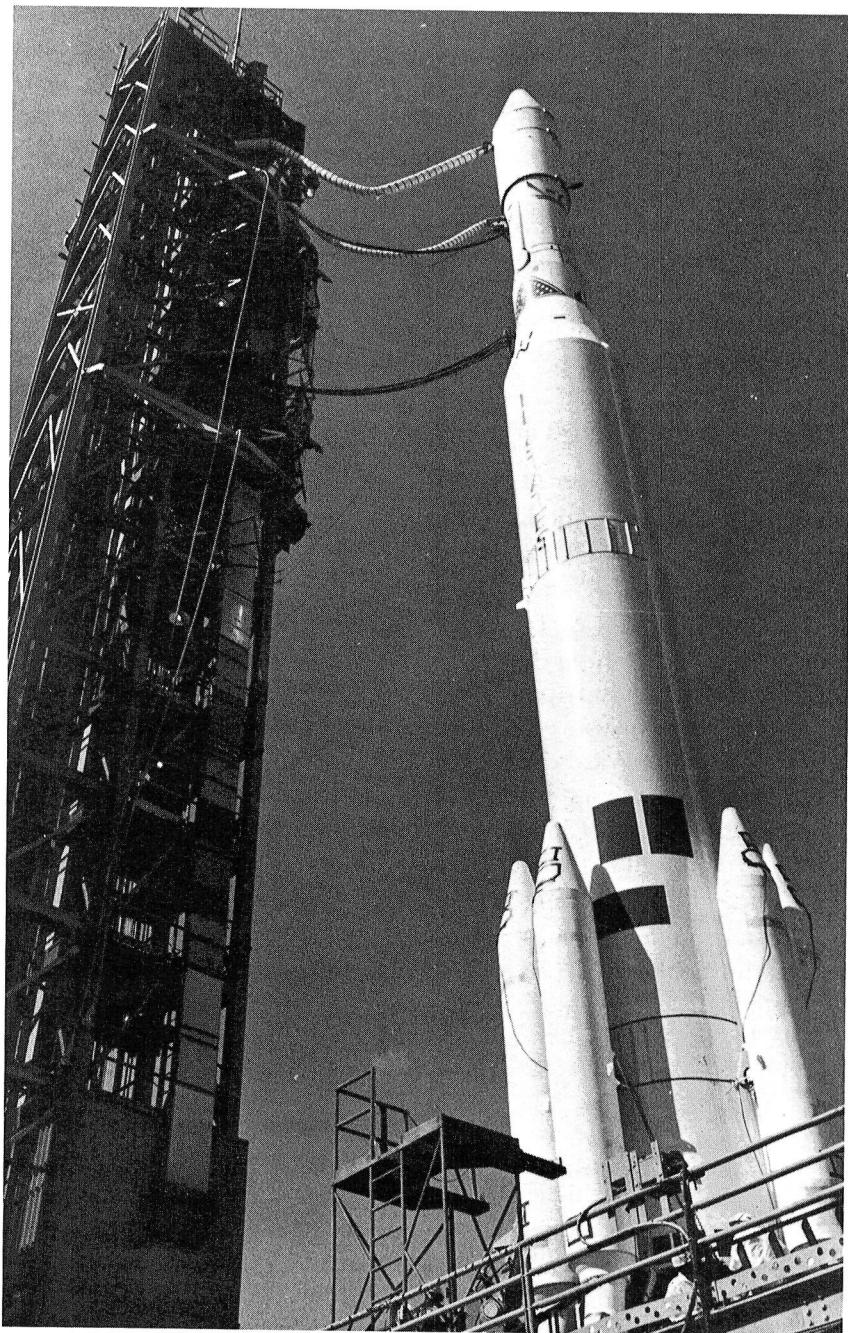


Figure 41. IMP I Spacecraft was Launched by a Delta Vehicle on March 13, 1971.

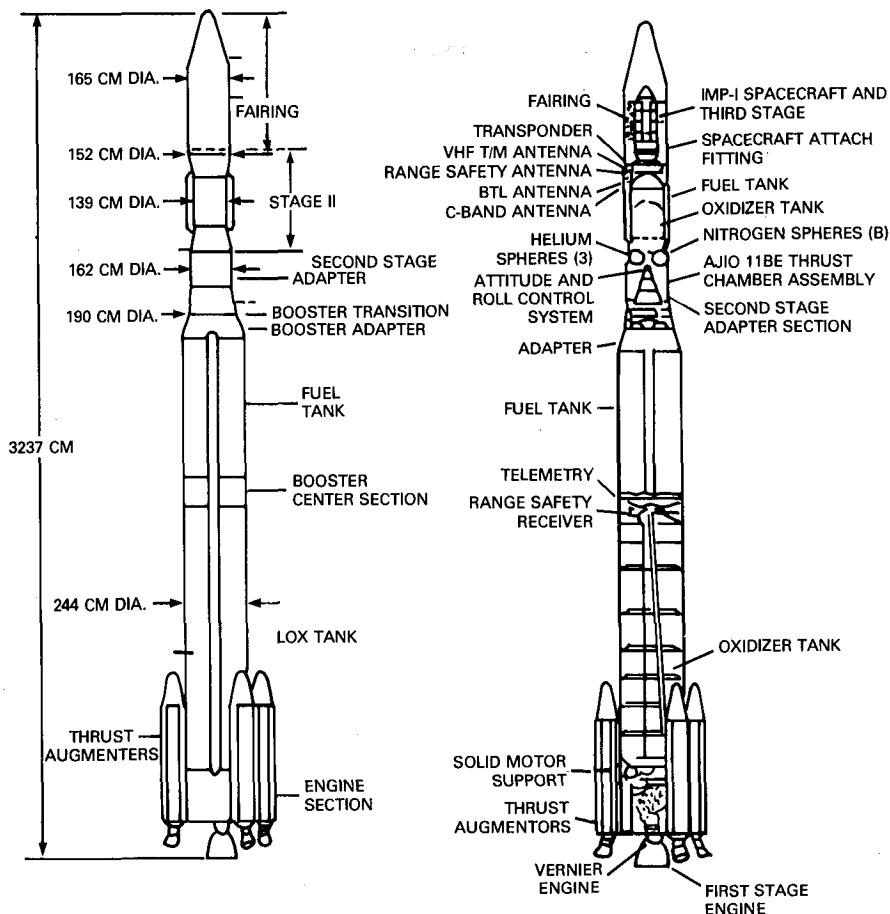


Figure 42. IMP I Launch Vehicle

Only spacecraft operations of major importance or ones dealing with failures are described below.

On March 31, 1971, the large telescope portion of the University of Chicago's Cosmic Ray Experiment failed. The failure was traced to the connection from the first sensor in the telescope to the amplifying electronics.

The high voltage power supply for sensor #2 of the GSFC Plasma Experiment failed in April 1971. One day later, the high voltage supply for sensor #1 failed; however, the rest of the electronics continued to function and only housekeeping data were telemetered. On November 17, 1971 the experiment was turned off.

The Cosmic Ray Experiment developed a turn-on problem on April 11, 1971. The problem occurred with the low energy detector (LED) of the experiment. This experiment was normally commanded off below 40,000 km. During the turn-on, the LED output was partially abnormal; i.e., pulse height accumulator data were not correct. It was found that by issuing a number of on-off commands, the system returned to normal. The number of commands needed, however, increased with time. On January 18, 1972, the system would no longer respond to commands. The LED was allowed to remain on above 40,000 km since the sector data were usable.

The "On-Board Computer" used in the engineering test operated intermittently for the first 40 days of flight because of a problem which caused the program to halt after several hours of operation. Considerable experimentation with the flight system and with an engineering model on the ground indicated that the problem was in the computer's priority interrupt hardware. A modified program eliminated the problem. It was then loaded and the computer operated normally. The on-board program was revised several times at the experimenter's request based on evaluation of actual flight data.

On October 14, 1971, IMP I entered the Earth's shadow at 1130 hours universal time (UT) and was predicted to exit at about 1708 hours UT. The exact duration of the shadow was unknown since the spacecraft transmitters were not commanded on until an hour after estimated shadow exit. For transit through the shadow, all systems were turned off except the solar proton experiment, the data handling system, and the command receivers. These systems remained on during the entire shadow period. It was estimated that the spacecraft temperature dropped to below -60°C during its time in the Earth's shadow.

During the long shadow period, the attitude control system leaked some of its Freon propellant. The leakage was due to the extremely low temperature that the valve seat was exposed to. This leakage caused the spacecraft spin rate to drop from 5.39 rpm to 4.76 rpm. At an experimenter's meeting, it was decided to allow the spin rate to remain at the lower value.

The spacecraft survived the long shadow period with minor degradation. The spacecraft radio frequency (RF) noise level increased slightly after exit from the shadow. The University of Chicago Cosmic Ray Experiment's main detector guard channel had an electronic failure. The main detector failed the second week in orbit but the guard channel was working. All other experiments were unaffected by the shadow.

After a short passage of the IMP I through a shadow on April 18, 1972, the University of Chicago's Cosmic Ray Experiment main telescope, which failed shortly after launch, returned to normal operation.

On September 26, 1972, one of the GSFC Radio Noise Experiment receivers failed and about the same time the GSFC Solar Proton Experiment failed to respond to turn-on commands.

After IMP I had been in orbit for 3 years and 3 months, ten of twelve experiments were operating with four out of the ten experiments degraded.

During the first week of October 1974, IMP I re-entered the Earth's atmosphere, bringing an end to its mission.

IMPs H AND J

In previous missions, IMPs A, B, and C carried out the exploration of the near-interplanetary region, the outer portions of the Earth's magnetosphere, and the interactions of the Earth-Sun system. AIMPS D and E were planned for lunar orbits. IMPs F and G studied the regions covered by IMPs A, B, and C during a period of maximum solar activity while IMP I continued previous studies, and also obtained electric field data.

The primary objective of IMPs H and J was to obtain a more detailed understanding of the regions broadly surveyed by each of the previous IMPs. Their nearly circular orbits were designed to continuously monitor the interplanetary medium and the geomagnetic tail plasma sheet throughout an entire year.

The specific purpose of the IMPs H and J spacecraft was to:

- Perform detailed and near-continuous studies of the interplanetary environment during several rotations of active solar regions; and
- Study particle and field interactions in the distant magnetotail including cross-sectional mapping of the tail and neutral sheet.

Two spacecraft were launched due to the desire for experimental measurements in orbits 180 degrees out of phase with each other so that correlative studies could be made. They represented a significant step forward in the evolution of Earth-orbiting IMPs. A sizeable increase in experiment weight, information rate, and available electrical power was now available.

Spacecraft Description

IMPs H and J (Figures 43 and 44) were essentially the same as IMP I. The overall height of the spacecraft was reduced by 24.6 cm, however.

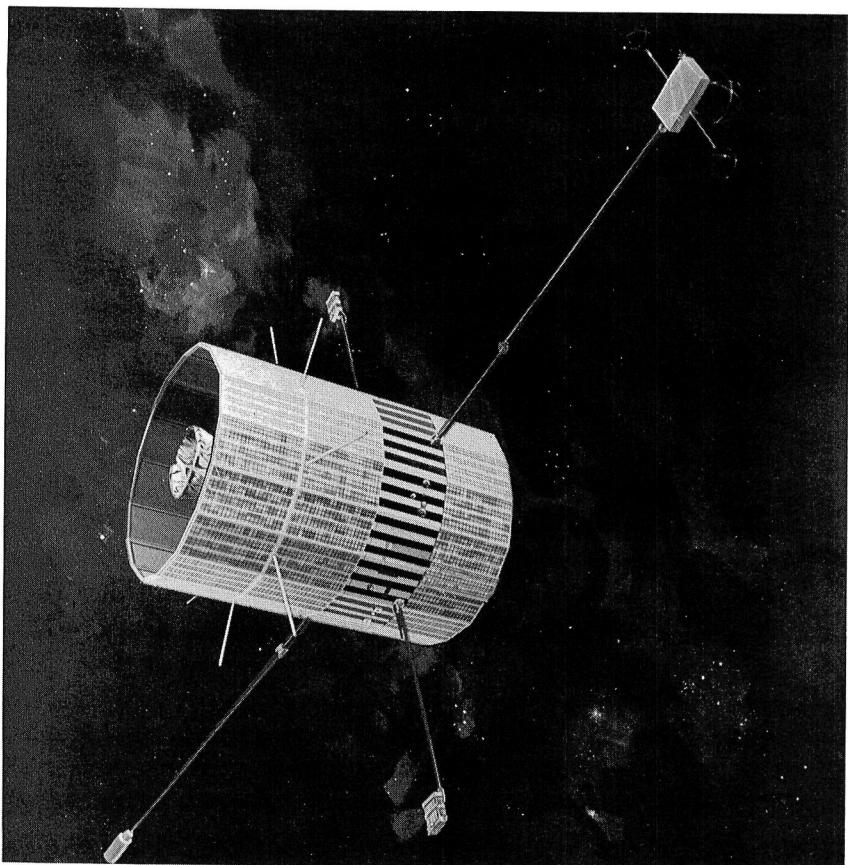


Figure 43. IMP H Spacecraft.

Also, the scientific experiment complement was increased from twelve to thirteen on IMP H and the on-board engineering test complement was increased from one test on IMP I to three on IMP H. IMP J carried twelve scientific experiments and two engineering tests.

IMPs H and J were structurally similar to IMP I. A kick motor was added and the structure was modified to make room for the experiments. Figure 45 shows the IMP H structure.

Appendages

IMP H appendages were essentially the same as those in the IMP I, except that changes were made in the length of some. In addition, the EFM antennas were not used on IMP H.

The two IMP H attitude control system booms were approximately 1.2 m, a reduction in length of 0.3 m from the booms used on the IMP I spacecraft. The two IMP H experiment booms were approximately 3 m in length, a reduction of 0.5 m from the booms used on the IMP I. Also, these booms were double-hinged on IMPs H and J, whereas they were triple-hinged on the IMP I.

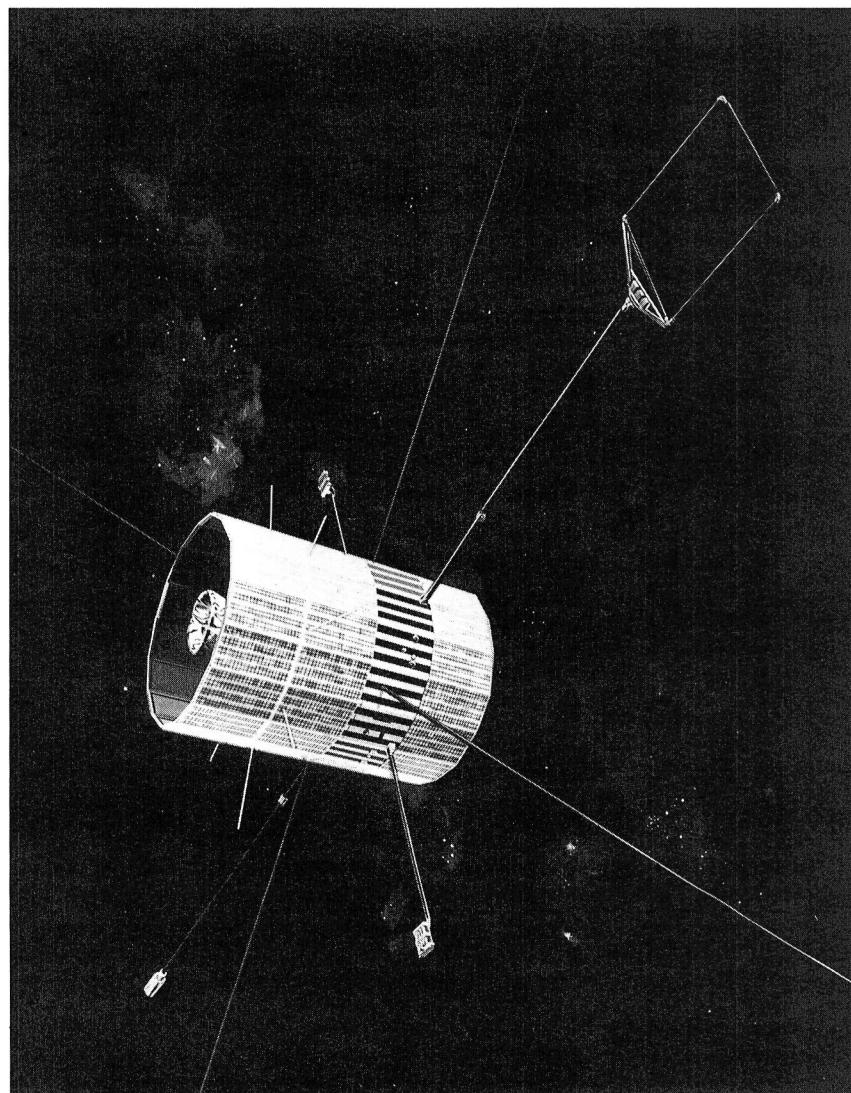


Figure 44. IMP J Spacecraft.

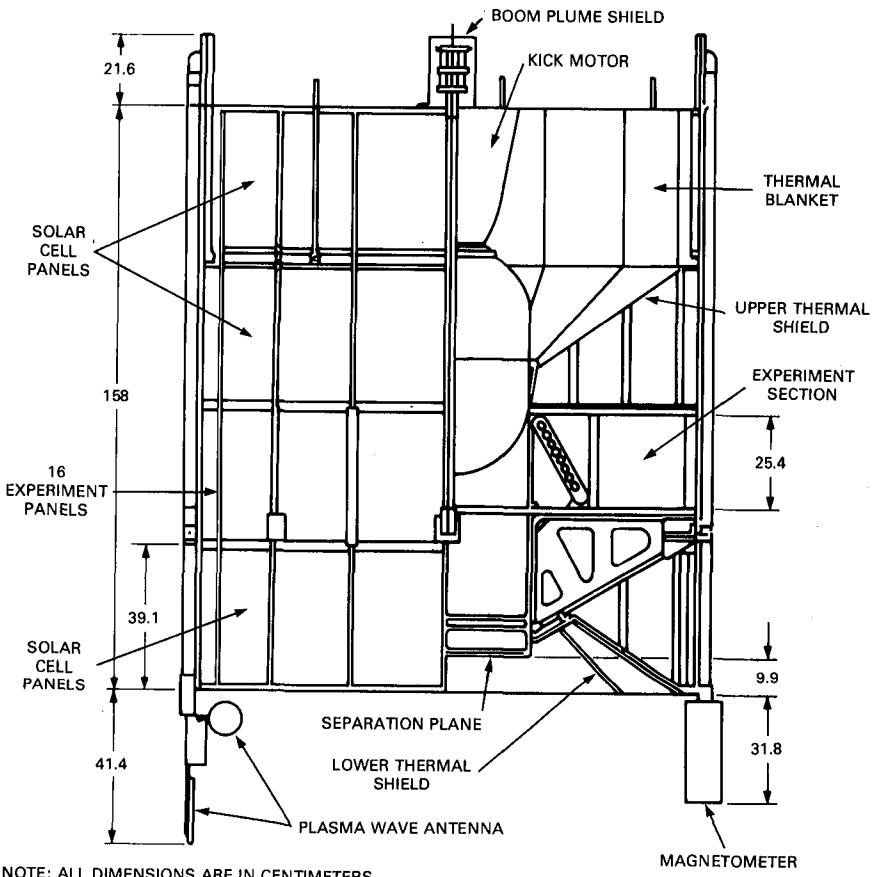


Figure 45. IMP H Structural Launch Configuration.

On IMP J, the length of the two experiment booms was increased slightly. Also, IMP J had four 61 m experiment antennas which were deployed after orbit had been achieved, and two inertia booms each 47 cm long.

Power Supply System

As in prior IMPs, IMPs H and J used body-mounted solar panels for energy conversion, a silver-cadmium battery for storage, and a solar array regulator for prevention of excess voltage.

A total of 3,264, 2 by 6 cm, N-on-P silicon solar cells were used to convert solar energy into electrical energy. Sixty-eight solar cells were mounted on each of 48 solar panels. Three rings each composed of 16 solar panels made up the external cylindrical surface of the spacecraft. Two of the solar rings containing solar cell panels were mounted forward and one aft of the experiment module section. The three rings of solar cells were designed to supply approximately 150 watts of power initially and 130 watts after 1 year, at 28 volts during normal operation.

Regulation was provided for three different modes:

- 1) When the solar array output was sufficient to support the load, a direct transfer of energy to the load occurred; as the load increased or the array output decreased, the battery was used to support the load;
- 2) When the load was less than the array capacity, the battery was recharged;
- 3) When the array output exceeded both the load and battery charge requirements, the excess was dissipated as heat; heat dissipation was accomplished by using a "dump circuit" made up of resistors and transistors.

The power distribution system provided 28 volts of power to the spacecraft subsystem and experiments. Converters transformed this voltage to appropriate levels for the various electronic equipment.

Communications and Data Handling System

The communications system, command system, range and range rate, antenna system, and telemetry and data system were essentially the same as those used on the IMP I spacecraft. Transmitters for IMPs H and J had increased power with 12 and 8 watts, instead of 8 and 6 watts used on the IMP I.

Antennas

The antenna system for IMPs H and J consisted of two diplexers, a hybrid circulator, and an eight-element monopole turnstile array (four active and four passive). The characteristics of the antenna system were as follows:

Frequency range	136-148 MHz
Polarization	Linear
Maximum gain (includes passive element losses of 1 dB)	+2 dB
Null depth	-6 dB based on station aspect angle

Transmitters

Two transmitters were also used on IMPs H and J. The primary transmitter operated at a frequency of 137.980 MHz, at 12 watts output. The secondary transmitter operated at a frequency of 136.800 MHz, with an output of 8 watts.

Receivers

As in IMPI, two telemetry command receivers were used, each operating at 148.890 MHz. One receiver received the encoded PCM/FSK commands and the RARR interrogation signals and provided a ranging signal for transmitter No. 2, a detected PM signal for the ranging decoder, and a detected AM signal for the PCM command decoder. The PCM command decoder had two outputs—one for 64 spacecraft commands, and the other for DST commands. The detected PCM/FSK spacecraft commands were used to back up 64 of the 125 tone-sequential spacecraft commands. The DST contained two 1024-bit format control memories which were loaded by the detected PCM/FSK DST commands.

Another receiver received the tone-sequential commands that were transmitted by the ground stations and provided a detected AM signal to the tone-sequential decoder. Each tone-sequential command was composed of a four-tone sequence, consisting of an address tone and three execute tones.

Telemetry and Data System

The telemetry system for IMPs H and J provided a means for transmitting PCM and analog telemetry data. The primary spacecraft-to-Earth link was provided by the 137.890 MHz, 12-watt signals from the primary transmitter. The carrier was phase-modulated and had a peak phase deviation of 1.1 radians. This transmitter was used exclusively for transmitting 800 or 3200 bits per second, prime PCM split-phase data. Convolutional coding with a one-half rate code (i.e., only one-half the bit rate was information, the other half was parity) was used.

The secondary spacecraft-to-Earth link was provided by the 136.800 MHz, 8-watt, phase modulated transmitter for transmitting RARR and the analog data.

Performance Parameters

IMP H had the capability of telemetering 104 performance parameters. However, only 85 of these were actually used which included 46 for analog and 39 for digital measurements.

On IMP J, 99 performance parameters were used. Of the 99 parameters, 51 digital parameters were measured and an additional five channels were spares. Forty-eight analog performance parameters were measured.

Thermal Control

IMP H thermal control was similar to that for IMP I where passive elements were used. However, some major changes in design were necessary. The principal component on the IMP H differing from IMP I was the fourth stage apogee kick motor. This motor was not ejected after firing; therefore, heat transferred back into the spacecraft after motor firing was a major consideration in the thermal design of spacecraft and motor.

The following additional thermal control techniques were used on IMP H:

- 1) A high temperature fiberglass isolator was placed between the motor flange and the upper center tube and between the center tube and the main equipment platform.
- 2) A 25-layer insulation blanket consisting of aluminized Kapton was placed around the motor and nozzle. Also, a one-layer shield of aluminized Kapton was placed over the nozzle exit plane. Thermostatically controlled heaters covered most of the surface area of the motor and nozzle and were divided into three separate sets, two for the motor and one for the nozzle, each individually controlled.
- 3) The upper part of the spacecraft and boom appendages were protected from motor plume heating with multi-layer insulation or thermal blankets and plume shields made from aluminized Kapton (Figure 45).
- 4) The IMP H battery, unlike the IMP I, was maintained at $15^{\circ}\text{C} \pm 5^{\circ}\text{C}$. The battery was thermally isolated (from heat conduction) from the platform. Because of this isolation, a 9-watt thermostatically controlled heater was added to maintain temperature.

Table 15 summarizes the thermal control techniques used on IMP H.

Kick Motor

Kick rocket motors made by Thiokol Chemical Corporation were used on IMPs H and J to inject the spacecraft into nearly circular Earth orbits. They were secured to the spacecraft structure (Figure 45) and ignited by a spacecraft command. The motor (Figure 46) provided a thrust of 1,543 kg and total impulse of 32,400 kilogram-seconds. The rocket

Table 15
Thermal Control Techniques Used on IMP H

External Surface	Technique
Experiment Panels	Black and White Paint
Upper Thermal Shield	Multilayer Insulation
Rear of Upper Solar Cell Panels	Multilayer Insulation
Fourth Stage Motor & Nozzle	Multilayer Insulation
Lower Center Tube Cover	Multilayer Insulation
Lower Thermal Shield	Black Paint
Lower Exposed Section of Center Tube	Buffed Aluminum
External Detector Housings	White Paint
Antennas	Black Paint
4 Booms	Aluminum Tape and Black Paint
ACS Nozzle Assemblies	Buffed Aluminum, Black and White Paints
Magnetometer Assembly	Vapor deposited aluminum, Buffed Aluminum, Black and White Paints
Plasma Wave Antenna	Multilayer Insulation and Black Paint

weighed 124 kg of which 112 kg was propellant. Provisions were made to telemeter motor chamber pressure, spacecraft acceleration, and motor temperature in real time by means of a Delta Instrumentation Package (DIP).

Spacecraft Launch, Orbit and Performance

IMP H

IMP H was launched on September 22, 1972 at 9:20 P.M. EDT from the Eastern Test Range, Cape Kennedy, Florida. The spacecraft was injected into a near-circular orbit 3 days later.

TE-M-521 ROCKET MOTOR

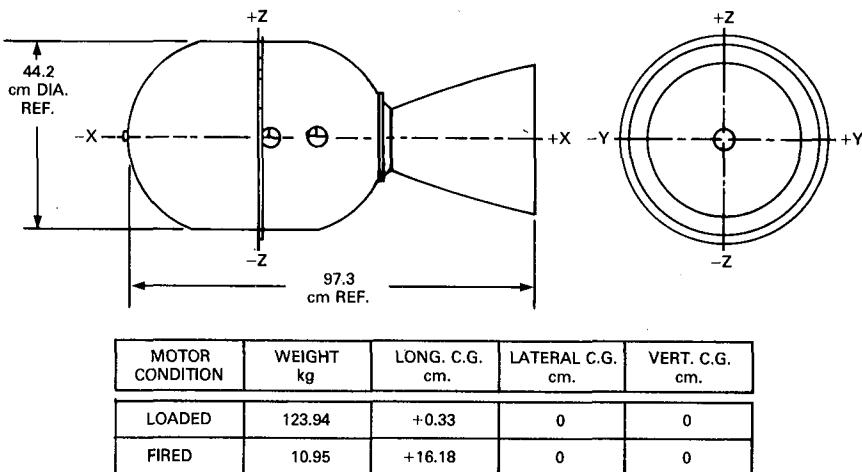


Figure 46. Kick Rocket Motor.

The Delta launch vehicle for the IMP H spacecraft was a Thrust Augmented Improved Delta 1604 three-stage rocket having an overall length of 32.3 m, and a maximum body diameter of 2.4 m. Figure 47 shows the vehicle configuration.

The first stage was a McDonnell Douglas Corporation modified Thor booster incorporating six strap-on solid-propellant rocket motors. The booster was powered by a Rocketdyne engine using liquid oxygen and hydrocarbon propellants. The engine was gimbal-mounted to provide pitch and yaw control from lift-off to main engine cutoff. Two liquid propellant vernier engines provided roll control throughout the first stage operation and pitch and yaw control from main engine cutoff to first stage separation.

The second stage was an Aerojet General Corporation engine employing liquid nitrogen tetroxide and Aerozene 50 as propellants. The second stage main engine was gimbal-mounted to provide pitch and yaw control through second stage burn. A nitrogen gas system using eight fixed nozzles provided roll control during powered and coast flight, as well as pitch and yaw control after second stage cutoff. Two fixed nozzles fed by the propellant tank helium pressurization system provided retro-thrust after third stage separation.

The third stage was a Thiokol Chemical Corporation spin-stabilized solid propellant motor which was secured to a spin table mounted to the second stage. Solid propellant rockets fixed to the spin table were used to spin-up the third stage assembly. The same standard Delta fairing as was used for IMP I was attached to the forward face of the second stage for IMP H.

Table 16 shows the initial values of the orbit achieved.

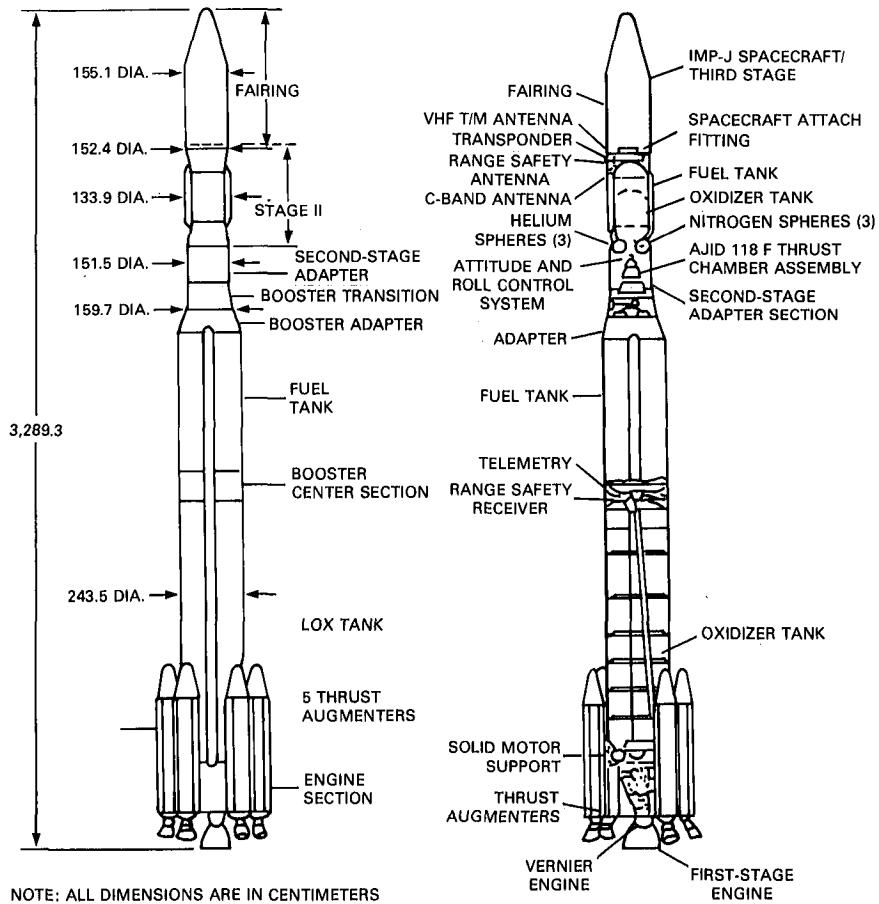


Figure 47. IMP H Launch Vehicle.

Table 16
IMP H Earth Orbit (Initial Values)

Apogee	235,609.41 km
Perigee	201,613.79 km
Inclination	17.21 degrees
Orbital period	12 days, 8 hours

Figure 48 shows the initial orbital path of the IMP H spacecraft. Only spacecraft operations of major importance, or ones dealing with failures, are described below.

On November 24, 1972, the proportional counter of the University of Maryland Ion and Electron Experiment failed to operate. The effect was a 10 percent reduction in the performance of the experiment. The counter was turned off on November 25. A month later, the mag-

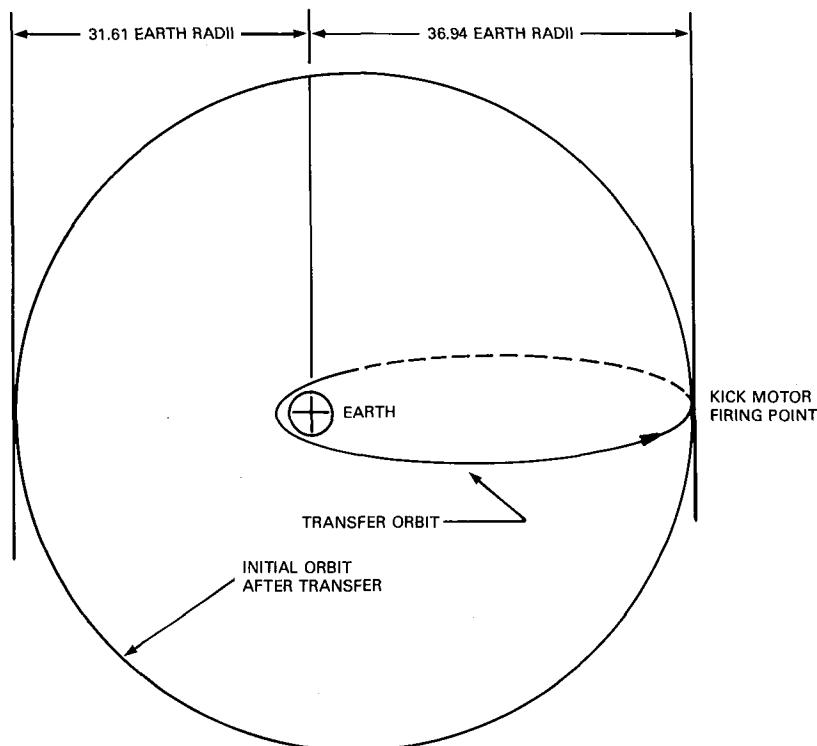


Figure 48. IMP H Initial Orbital Path.

netometer flipper mechanism for the GSFC Magnetic Field Experiment failed.

In January 1973, two of the three Geiger-Müller tubes in the Johns Hopkins University/Applied Physics Laboratory Particles Experiment failed. A third Geiger-Müller tube failed a month later.

On April 4, 1973, the GSFC Magnetic Fields Experiment failed to operate properly and was turned off.

IMP H experienced its first extended period in the Earth's shadow on April 9, 1973. The shadow period began at 7:30 P.M. UT and lasted for over 3 hours. Spacecraft shutdown began at 6:30 P.M. UT and concluded 15 minutes into the shadow. Experiment turn-on began at 1:30 P.M. UT on April 10, 1973, and was completed on April 11th. No problems occurred resulting from the spacecraft's passage through the shadow.

As of September 30, 1978, after over 6 years in orbit, data acquisition from IMP H was discontinued. Twelve of the 13 IMP H experiments remain operational on a standby basis. Of the 12, four were degraded.

IMP J

The IMP J was launched on October 25, 1973 at 10:20 P.M. EDT from Cape Kennedy, Florida (Figure 49). The same launch vehicle configuration used to launch IMP H was used for IMP J. Three days after launch, the kick motor was ignited and the spacecraft was injected into the mission orbit.

Table 17 shows the initial orbital parameters for IMP J. The initial phase angle with the IMP H spacecraft was 180 degrees as planned.

Table 17
IMP J Earth Orbit (Initial Values)

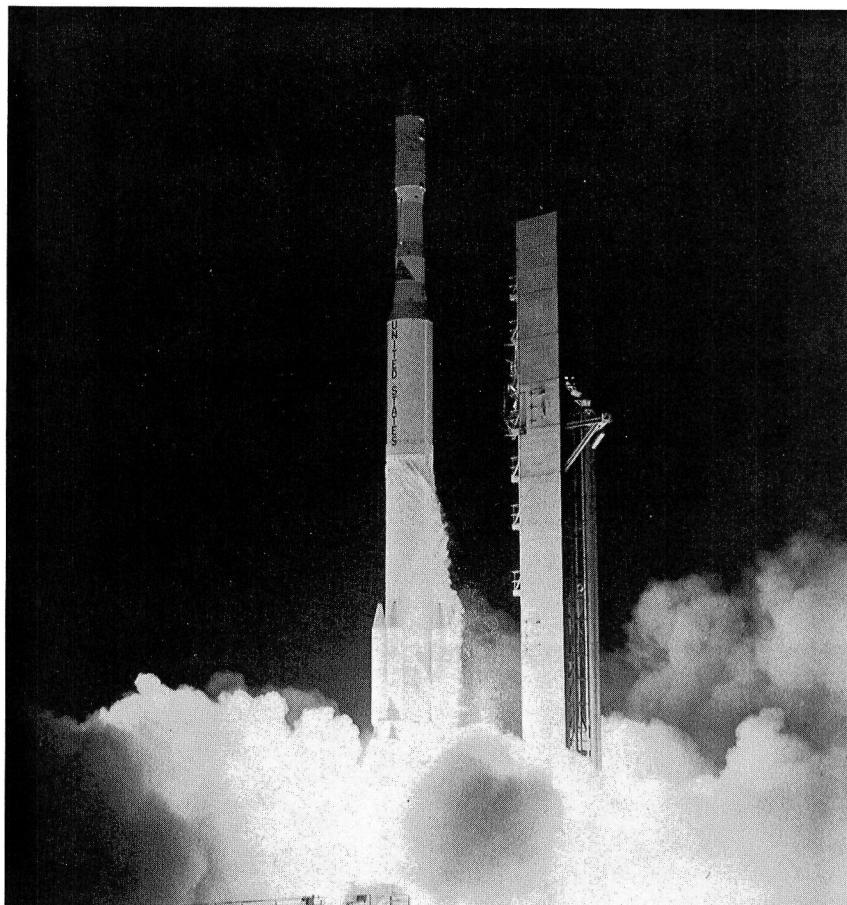
Apogee	288,672 km
Perigee	141,050 km
Inclination	28.64 degrees
Orbital period	11 days, 23 hours, 40.5 minutes

A maneuver to rotate the spacecraft 180 degrees was successfully accomplished on December 4, 1973. The spacecraft positive spin axis was placed within 2 degrees of being perpendicular to the ecliptic plane and pointing toward the north ecliptic pole.

On December 18, 1973, the y-axis electrical field measurement antennas were extended to 61 m each. The +x-axis antenna would not extend beyond 1.8 m, however, and several unsuccessful attempts were made to retract and deploy this element. The -x-axis antenna was deployed to 12 m and then was retracted to 1.8 m. Both electrical field measurement experimenters felt they could obtain almost all of their data using the y-axis dipole. The loss of the x-axis dipole amounted to a loss of redundancy.

As of the date of the publication of this book, the IMP J spacecraft, which has been in orbit for over 6 years, remains operational and experiment data collection is continuing.

Figure 49. The IMP J, the Last Spacecraft in the IMP Series, was Launched by the Delta Vehicle October 25, 1973.



Chapter 4

IMP Scientific Experiments

The scientific objective of the IMP missions was to advance significantly the knowledge of interplanetary space and its phenomena by means of direct measurements from orbits around the Earth and the Moon. In developing the spacecraft mission, particular emphasis was placed on four major study areas:

- Study the radiation environment in the Earth's magnetosphere and in cislunar space, and monitor the radiation in space in support of the Apollo program;
- Study the quiescent properties of the interplanetary magnetic fields and their relationship with the particle fluxes emanating from the Sun;
- Develop a capability for predicting periods of solar-flare activities; and
- Study solar-terrestrial relationships.

To meet these objectives, the NASA Space Science Steering Committee selected a number of scientific experiments which were carried into space by the series of IMP spacecraft over the ten-year period. These experiments were designed to measure magnetic fields, electric fields (IMPs I, H and J only), plasma, and energetic particles during the lifetime of the spacecraft flight. Several other experiments were developed and flown in the IMP series which included the IMP E micro-meteorite experiment, the AIMPs D and E passive lunar experiments, and the IMP I, H, and J radio astronomy experiments.

In addition to the scientific experiments, several engineering experiments were flown. The IMP spacecraft provided a "test bed" for advancing or proving spacecraft technology for future flight missions. The experiments carried on each spacecraft series, IMPs A, B, and C, AIMPs D and E, IMPs F and G, and IMPs I, H and J are described in the following paragraphs in order of their flight mission and phenomena measured.

IMPs A, B, AND C

Magnetic Field Measurements

Magnetometer Experiment (Goddard Space Flight Center—Dr. N. Ness)

This experiment, developed to measure magnetic fields, was flown on IMPs A, B, and C. It consisted of two different types of instruments—a Rubidium (Rb) vapor magnetometer and two fluxgate magnetometers.

The Rb-vapor magnetometer measured to total vector magnetic field independent of orientation. The magnetometer design utilized an optical pumping technique, creating resonance absorption to detect the magnitude of Zeeman splitting in the energy levels of rubidium. Since the Zeeman splitting appeared as a Larmor precession, detection was accomplished by measuring the Larmor-precession frequency. The Rb vapor magnetometer was used to calibrate the fluxgates but did not produce an independently useful data set. Two saturable-core fluxgate magnetometers were used to determine the direction of fields in regions where the field strength was less than 30 gammas (1 gamma = 10^{-5} gauss). Each fluxgate magnetometer measured by the component of the magnetic field along the sensor axis by detecting the second harmonic content in the secondary of the sensor transformer. The fluxgates functioned normally throughout the useful life of the satellite and provided usable data through May 1964 for IMP A and into April 1965 for IMP B. However, on IMP C, one fluxgate failed at launch, but the other performed normally, sampling the magnetic field 30 times within each of six 4.8-second intervals every 5.46 minutes. Useful fluxgate data were transmitted until May 11, 1967.

Cosmic Ray Measurements

Range Versus Energy Loss (University of Chicago—Dr. J. A. Simpson)

A charged particle solid-state telescope was used to measure range and energy loss of galactic and solar cosmic rays. The apparatus was designed to search for solar-proton or alpha-flare events should they occur while the satellite was transmitting. Proton energies between 100 kev and 200 Mev were to be measured with these instruments. The range-energy loss telescope used between 25 and 40 solid-state detectors, along with a 64-channel pulse-height analyzer and range logic similar to that flown in Mariner A.

From IMP A launch until October 15, 1964, a malfunction limited alpha studies. No useful information was received after that date. With the instrument on IMP B, useful data was obtained from launch until April 5, 1965. Data coverage was intermittent throughout the life of the spacecraft due to frequent spacecraft shutoffs and sporadic failure of some detectors. The experiment flown on IMP C performed normally until April 21, 1966, after which several problems with the instrumentation developed, causing spikes in the count rate data, especially in the lowest energy channel. The date of transmission of the last useful information was April 29, 1967.

Energy Versus Energy Loss (Goddard Space Flight Center-Dr. F. B. McDonald)

This experiment consisted of two detector systems. The first was a dE/dx vs. E telescope with thin and thick CSI scintillators (one each) and an anticoincidence plastic scintillation counter. The telescope axis was normal to the spacecraft spin axis. This experiment furnished precision separation of protons, electrons, alpha particles, and heavy primaries, and was sensitive down to very small flux values. This provided a means of determining energy and charge spectra. Proton and alpha-energy sensitivity covered the regions from 10 to 100 Mev/nucleon. It also provided mass separation of singly charged particles.

The second detector system consisted of two Geiger-Müller tube telescopes oriented parallel and perpendicular to the spacecraft spin axis. Each telescope consisted of two colinear GM tubes. The parallel and perpendicular telescopes furnished information on the isotropy of solar-proton events and of cosmic-ray modulation. Both detector systems worked well from launch until May 8, 1964 on IMP A and until March 1965 for IMP B. The IMP C experiment worked well until May 11, 1967.

Ion Chamber and Geiger-Müller Counter Tubes (University of California-Dr. K. Anderson)

The instrumentation for this experiment, designed to measure fluxes of geomagnetically trapped particles, consisted of a 3-inch diameter Neher type ionization chamber and two Anton 223 Geiger-Müller tubes. The ion chamber responded to electrons and protons. This instrument measured total ionization produced per unit time in a unit volume of standard-density air. It was simple to operate, maintaining a constant

calibration for extended periods of time, and was intended to serve as a basic radiation monitor. A gold foil scattering unit with the Anton No. 223 thin-window Geiger-Müller tube was used to detect electron fluxes above 45 kev. The background counter was also an Anton No. 223 tube. Both counters were surrounded by 0.318 cm copper shielding.

The IMP A experiment performed normally from launch through May 10, 1965. The IMP B experiments last day of transmission was October 13, 1965, while the IMP C experiment continued from launch until May 11, 1967.

Solar Wind Experiments

Three solar wind experiments were flown on IMP A, B, and C. These included a Low Energy Proton Analyzer, a Plasma Probe and a Thermal Ion and Electron Sensor. Each of these is discussed below.

Low Energy Proton Analyzer (Ames Research Center-Dr. J. H. Wolfe)

A quadrисpherical electrostatic analyzer with a current collector and an electrometer amplifier was used to detect and analyze the positive ion component of the incident plasma and to study its gross flow characteristics. Protons were analyzed in 14 energy channels. Proton concentrations, as a function of kinetic energy, were determined by admitting the protons through a slit of known dimensions in the satellite skin. A variable curved-plate electrostatic analyzer separated the particles according to their energy, producing a particle current which was a function of the energy level and could be measured by an electrometer circuit. By proper calibration of the analyzer in the laboratory, and given the geometrical and electrical characteristics, the particle concentration outside the satellite could be determined.

It was planned to maintain resolution down to 200 EV and provide an order-of-magnitude answer on concentrations at energies below this level. This effectively dictated a 20-keV upper limit, which was the highest energy level expected from the solar-wind protons.

No data were obtained for the brief periods when the IMP A satellite was in the magnetosphere. The instrument operated well until April 1964 when it started operating intermittently. Its operation continued to degrade thereafter. The IMP B experiment operated well during the time when data could be recorded. The IMP C instrument failed at launch and thus produced no useful data.

Plasma Probe (Massachusetts Institute of Technology-Dr. H. Bridge)

A five-element split collector Faraday cup was used to measure the flux of low-energy positive particles in particle-velocity increments between 10 km/sec and 1000 km/sec. Charged particles entering a 15.2-cm diameter surface area passed through a series of grids set at potentials which rejected electrons and low-energy positive particles. The voltage on one grid was modulated at potentials between 5 and 3,000 volts. The number of charged particles reaching the collector was detected with an electrometer circuit.

With the instrument on IMP A, useful data were obtained from launch until January 13, 1965. However, there was poor data coverage during the last 7 months because of intermittent satellite transmission. The IMP B instrument produced data throughout the operational life of the spacecraft, and provided essentially continuous data through April 5, 1965. The IMP C instrument failed at launch.

Thermal Ion and Electron Sensor (Goddard Space Flight Center-R. Bourdeau, G. Serbu)

The retarding potential analyzer was a three-element planar Faraday cup. It was mounted normal to the spacecraft spin axis. This experiment measured the concentration and temperature of thermal electrons (energies less than a few electron volts), and the concentration, masses, and temperatures of thermal ions over the entire orbit. One sensor with different modes of operation was used. The sensor was a charged-particle trap, similar to the ones flown on Explorer 8 with three electrodes (two grids and one collector).

On IMP A, the experiment operated from launch for about 20 hours when failure of a mechanical programmer switch terminated operations. The data were adversely affected by secondary electrons.

On IMP B, the instrument experienced secondary electron contamination but returned essentially continuous data until April 5, 1965. The same electron contamination was experienced on IMP C but it operated the entire life of the spacecraft until May 11, 1967.

AIMPs D AND E**Passive Experiments**

Two passive experiments were flown on AIMPs D and E which would use data transmitted from the spacecraft for other purposes while obtaining valuable lunar information.

Lunar Ionosphere Effects on Radio Wave Propagation (Stanford University–A.M. Peterson)

The purpose of this experiment, which used a Cislunar RF Beacon, was to study the electromagnetic reflective properties of the lunar surface using the satellite telemetry signal. However, because AIMPS D did not achieve lunar orbit, the experiment could not be performed. On AIMPS E the experiment operated normally during the spacecraft lifetime.

Selenodetic Information (University of California–W.M. Kaula)

Range and range-rate tracking data was to be used to obtain selenodetic information. However, because AIMPS D did not achieve lunar orbit, the experiment could not be performed. On AIMPS E, information was obtained during the spacecraft lifetime.

Magnetic Field Measurements***Ames Magnetometer Experiment (NASA/Ames Research Center–C.P. Sonnett)***

The Ames Magnetometer Experiment consisted of a boom-mounted sensor. It was located approximately 2 m from the spacecraft and was perpendicular to the spin-axis. The sensor unit consisted of three orthogonally mounted individual fluxgate sensors (x, y and z). Sensors x and y were mounted perpendicular to the spin-axis, and sensor z was mounted parallel to the spin-axis. A flipper rotated sensors x and z 90 degrees once each day in order to orient the sensor's positions for calibration. The system was designed to measure the spatial and temporal variations of the interplanetary and lunar magnetic fields in three linear ranges.

On AIMPS D and E, the instrument performance was normal until the final spacecraft transmission.

Goddard Magnetometer (Goddard Space Flight Center–N.F. Ness)

The Magnetometer Experiment was a boom-mounted, three-component, fluxgate magnetometer equipped with a flipper mechanism. The orthogonally mounted sensors were arranged with one sensor parallel to the spin-axis of the spacecraft and the remaining two perpendicular to the spin-axis. The flipper mechanism reoriented the sensor array once every 24 hours by rotating one perpendicular sensor and the parallel-mounted sensor 90 degrees around the axis of the remaining

sensor; this procedure achieved the same effect for calibration purposes as flipping the spin-axis sensor 180 degrees.

On AIMPE, the detector functioned well between launch and October 10, 1968, but it provided no useful data after that date. The AIMPE experiment performed normally during the lifetime of the spacecraft.

Plasma Experiment

Plasma Probe (Massachusetts Institute of Technology-H.S. Bridge)

A split-collector Faraday cup mounted on the spacecraft equator was used to study the directional intensity of solar wind ions and electrons. This experiment was designed to measure: angular distribution of the total proton flux in the equatorial meridian plane of the spacecraft; energy distribution of the proton flux at or near the same angle as a peak of the total proton flux; angular distribution of the electron flux in the equatorial and meridian plane of the spacecraft; and the energy distribution of the electron flux at or near the same angle at the peak of the total electron flux.

The experiment on AIMPE worked well from launch until final spacecraft data transmission on May 31, 1971, whereas the experiment on AIMPE failed in July 1968.

Cosmic Ray Experiments

Ion Chamber (University of California-K.A. Anderson)

The purpose of this experiment was to measure energetic particles and search for low energy solar electrons in interplanetary space. The main interest was in the electronic fluxes inside of gas clouds which produce magnetic storms. The ion chamber provided a precise and sensitive monitor of changes in the galactic cosmic-ray intensity and provided a time history of solar cosmic-ray events. The ion chamber on AIMPE operated normally from launch through September 2, 1966. Between September 2, 1966 and October 20, 1967, the date of last usable data, the ion chamber operated at a lower threshold voltage.

On AIMPE this experiment performed well initially but in November 1968, the ion chamber failed.

Electrons and Protons Experiment (University of Iowa-J.A. Van Allen)

The objectives of the Electrons and Protons Experiment were to study spatial, temporal and angular distributions of electrons with energies

exceeding 50 keV in the magnetospheric wake of the Earth at 60 Earth radii, to search for electrons with energies exceeding 50 keV in the wake of the Moon, and to conduct a detailed study of their distribution; in addition, to study the incidence and intensity of low energy solar cosmic rays versus time profile in interplanetary space, determine their energy spectra and angular distribution, and study solar X-rays in the 0-14 angstroms range. In the experiment on AIMP D, an intermittent, recognizable electronic failure occurred in September, 1966. Accumulator failures occurred in July, 1967, and September, 1967. A limited amount of usable data was collected through the date of final spacecraft transmission which occurred May 31, 1971. The AIMP E experiment collected data during the life of the spacecraft.

Low Energy Integral Spectrum Measurement (Thermal Ion and Electron) Experiment (Goddard Space Flight Center—G.P. Serbu and E.J.R. Maier)

This experiment was designed to measure low energy electrons and ions in the vicinity of the Moon, and to detect the presence or absence of a magnetosheath and/or shock-front associated with the Moon. In addition, the experiment was to observe the flow of the solar wind around the Moon and determine if the observed electron high-energy component as measured by an integral spectrum experiment can be interpreted in terms of solar wind electron temperature and number density by comparison with data from the plasma probe. On AIMP D, the experiment operated until the end of June 1967, while the AIMP E experiment operated over the spacecraft lifetime.

Micrometeorite Flux Experiment (Temple University—J.L. Bohn and W.M. Alexander)

This experiment measured the momentum, kinetic energy, and velocity of individual cosmic dust particles using film charged detection, induction devices and microphones. The information obtained was used in calculations involving perturbations of dust-particle distributions in cislunar space; distributions and sources (Moon or deep space) or dust particles in the vicinity of the Moon; dust particles associated with meteor streams; the nature of dust-particle mass distributions in interplanetary space; and effects of the geomagnetic tail and wake on dust-particle distributions and directions at lunar distances. The experiment on AIMP E operated normally from the time of launch through the life of the spacecraft.

Engineering Tests

Two engineering tests were carried aboard the AIMPE (in addition to the attitude control system, which was also considered an engineering test). One of these, the solar cell damage test, was essentially the same as that used on the AIMPD satellite. The additional engineering test consisted of a contamination monitoring experiment by R. Sheehy, GSFC. The purpose of the monitor was to determine the source of the contamination that degraded the thermal coatings on the top cover of the AIMPD satellite.

IMPs F AND G

Engineering Tests

RADEM Monitor (Goddard Space Flight Center-Mr. J.L. Wolfgang, Jr.)

The RADEM (radiation damage effect on MOSFETS) experiment flown on IMPs F and G was designed to monitor important electrical parameters of the metal oxide silicon field effects transistors (MOSFETS) devices to determine quantitatively the effect of exposure to the radiation environment in space. Since the spacecraft carried a great number of MOSFETS in its information processing subsystems, a controlled test of this type was considered necessary.

Measurements were made of the changes in the gate threshold voltage of the MOSFETS under controlled radiation shielding of two grams and one gram per square centimeter and with no shielding.

Another parameter monitored was the leakage between drain and source when the MOSFETS were turned off. It was known that p-channel-enhancement-type MOSFETS were subject to developing leakage after being irradiated, and that the effects were more pronounced in the off-state of the MOSFETS.

Scientific Experiments

Low Energy Telescope (Bell Telephone Laboratories-Dr. W.L. Brown)

Using a four-element, solid-state telescope, this experiment was to measure the energy spectra of relatively low energy electrons, protons, deuterons, tritons, and alpha particles outside the Earth's magnetosphere and in the region of the magnetospheric boundary. The experiment of IMP F performed normally from launch to spacecraft reentry date of May 3, 1969. The IMP G instrument failed in April 1971.

Ion Chamber (University of California (Berkeley)–Dr. K. Anderson)

This experiment, which was also used on earlier spacecraft, greatly advanced the description of energetic particle populations in and beyond the Earth's magnetosphere, and knowledge concerning dynamic processes that influence these populations and their relationships to solar phenomena. The experiment involved measurements made with a Neher-type ion chamber which provided an immediate and direct way of determining ionization produced by a spectrum of particles in matter.

The IMP F instrument performed normally from launch through May 3, 1969, when IMP F reentered the Earth's atmosphere. On IMP G, the ionization chambers began to stop intermittently in March 1971.

Range Versus Energy (University of Chicago–Dr. J.A. Simpson)

This experiment was essentially the same solid state telescope as the one used on the IMPs A, B, and C and was part of a long-range study of the spectra of low-energy galactic and solar particles in the period when solar activity began to increase from the solar minimum of 1964-65.

Scientific objectives included measurements of the isotope ratios of hydrogen and helium on the basis of experience with the IMP C spacecraft telescope; and further increasing the understanding of the modulation of galactic cosmic rays taking place beyond the Earth's orbit. The experiment also studied the changes of the fluxes and spectra of protons, alpha particles, and heavier nuclei for the scientific lifetime of the spacecraft. Except for the failure of the electron detector 6 days after launch, the IMP F experiment performed normally until the satellite decayed on May 3, 1969. The IMP G experiment also performed normally through the life of the IMP G mission.

Low-Energy Proton and Electron Differential Energy Analyzer (University of Iowa–Drs. J.A. Van Allen and L.A. Frank)

This experiment was designed to survey the spatial distribution of electrons and proton in the energy range of 100 EV to 50,000 EV inside the Earth's magnetosphere, in the transition region adjacent to the magnetospheric boundary, and in the interplanetary medium in the vicinity of the Earth's magnetosphere. The instrument also conducted a temporal study of these spatial distributions, making a comprehensive survey of the differential energy spectra of electrons and protons within the magnetosphere and in the transition region. The instruments on IMP F and G performed normally from launch until the satellites decayed.

Cosmic Ray Anisotropy (Southwest Center for Advanced Studies—Dr. K.G. McCracken)

This experiment was designed to study the degree of anisotropy of the low energy portion of the solar cosmic radiation, and determine the manner in which it varied with time and nuclear species. In particular, the experiment measured the anisotropics in the solar cosmic radiation with an angular resolution of 45 degrees or better as a function of time and as a function of energy, for both protons and alpha particles, from a point far away from the magnetosphere. On IMP F and G, the proportional counter and telescope worked well from launch until the spacecraft reentry dates.

Spherical Electrostatic Analyzer (TRW Systems Group, TRW Inc.—Dr. F.B. Harrison)

The spherical analyzer was designed to study the directional properties, absolute intensities, time variations, and energy spectra of protons, electrons, and (where present) alpha particles in the energy range below 10 keV. The prime objective was to gather data pertaining to a detailed energy spectrum of the solar wind. A second objective was to determine the ratio of protons to alpha particles in the solar wind, and study the directional properties of the solar wind. This experiment failed to operate on IMP F and was not flown on IMP G.

Solar Proton Monitoring Experiment (Johns Hopkins University/Applied Physics Laboratory Dr. C. Brostrom; Goddard Space Flight Center—Drs. D.J. Williams, D.E. Guss, K.W. Olgilvie)

This experiment was a joint venture of GSFC and the Johns Hopkins University Applied Physics Laboratory. The experiment consisted of an array of four solid-state detectors designed to measure proton intensities in the following energy ranges: $E > 10$, 30 and 60 MEV. Separate detectors were used for each energy range. The method employed allowed for accurate, absolute flux determinations and an accurate unit-to-unit comparison. On IMP F, data was obtained from the first three detectors between launch and May 3, 1969. Data from the fourth detector were obtained between launch and June 12, 1968. On IMP G all the detectors functioned normally through the lifetime of the spacecraft.

Plasma Experiment (Goddard Space Flight Center—Dr. K.W. Ogilvie; University of Maryland—Dr. T.D. Wilkerson)

An electrostatic analyzer and a velocity selector were used for this experiment which was to determine the composition and energy distribution of ions in the interplanetary plasma. The scientific objectives were to determine the relative abundance of hydrogen (H+) and Helium (He++) ions in the solar wind and transition region; to study flux and energy spectra variations for H+ and He++ ions in the solar wind and magnetosphere; and to study ion fluctuations over periods of 3 seconds and more. The instrument on IMP F operated normally until January 30, 1968. The instrument on IMP G operated intermittently through July 1969, and provided no data beyond that date.

Low Energy Proton and Alpha Detector (Goddard Space Flight Center—Dr. D.E. Hagge)

The scientific objectives of this detector were to measure low-energy cosmic ray proton flux in the 0.4 to 8 MEV range; to study the low-energy galactic cosmic ray alpha flux in the 2 to 8 MEV/nucleon range; to study the complete time history of protons and alphas in these ranges throughout solar cosmic ray events; and to study further the possible anomalous emission or storage of low energy solar particles as observed in the recurrent events detected by earlier Explorer satellites. The experiment collected data during the life of the spacecraft.

Energy Versus Energy Loss (Goddard Space Flight Center—Dr. F.B. McDonald)

This experiment used the E-versus-de/dx telescope to measure flux and energy spectra of hydrogen, deuterium, tritium and helium in the primary and solar cosmic radiation between 12 and 30 MEV/nucleon. Also, the flux and energy spectra of electrons in the 1 to 20 MEV range were measured.

Scientific objectives were to obtain a precise determination of the quiet-time galactic hydrogen and helium spectra in the 12 to 80 MEV/nucleon interval and the modulation of these components as a function of time; to study the dynamics of solar cosmic ray events as observed with two components of different charge-to-mass ratios (hydrogen and helium); to study the intensity of modulation processes involving electrons of 1 to 20 MEV; and to study the isotopic abundance of deuterium and tritium in the primary and solar cosmic radiation. The experiment functioned normally during the life of the IMP F and G spacecraft.

Magnetic Field Experiment (Goddard Space Flight Center-Dr. N.F. Ness)

This boom-mounted magnetometer experiment was designed to measure magnetic field vectors in space with high accuracy and precision. Three major subjects for investigation were: the interplanetary magnetic field; the magnetic field in the magnetosheath; and the magnetic tail field of the Earth. The fluxgate magnetometers simultaneously measured the relative magnetic field intensity along three mutually orthogonal axes. The experiment performed well during the entire lifetime of the IMP F and G missions.

Low Energy Solar Flare Electron Detector (University of California (Berkeley)-Dr. R. Lin)

This experiment which flew on IMP G only measured solar flare electron fluxes in the energy range of from 2.5 to 7.5 keV and 7.5 to 12.5 keV. Its purpose was to complement the University of California experiment on board the IMP G spacecraft which measured electrons in the following energy ranges: >15 keV, >45 keV, >80 keV and >160 keV. Measurements in the low ranges of 2.5 to 12.5 keV were important in positively linking solar radiation emission to energetic electrons and their release by the Sun. Due to high background count rates, only data of low quality were obtained. On December 2, 1969, the experiment was turned off.

Low Energy Proton Differential Energy Analyzer (LEPDEA) (University of Iowa-Dr. L.A. Frank)

A low-energy particle detector was employed to measure the differential energy spectra and angular distribution of low-energy positive ions over the energy range of 90 EV to 12 keV to provide definitive observations of positive-ion intensities in the solar wind, within the magnetosheath, and in the geomagnetic tail. This experiment was designed to continue monitoring the solar wind to determine positive-ion differential energy spectra over the 90 EV to 12 keV energy range; to determine the angular distribution of positive-ion intensities at various angles near the spacecraft-to-Sun direction. It was also designed to survey and monitor temporal variations of the solar wind positive-ion mean velocity, direction of arrival and temperature and determine the angular distributions of positive-ion intensities within the above-listed band passes at 15 angles about the spacecraft spin axis. The experiment performed normally for about two and one half months from launch when the experiment power supply failed.

IMP I**Energetic Particle Experiments*****Cosmic Ray Experiment (Goddard Space Flight Center–Dr. F.B. McDonald)***

The Cosmic Ray Experiment was designed to measure charged spectra, compositions, and angular distributions in the range of 0.5 to 500 MEV/nucleon for electrons, protons, and heavier particles up to $Z \cong 30$. The experiment consisted of a coordinated set of three charged-particle telescopes to measure flow patterns of energetic solar and galactic particles separately in the interplanetary field and to measure the energy spectra and isotropic compositions of galactic and solar cosmic rays from the lowest practical energies up to 500 MEV/nucleon. It also measured the time variations of the differential energy spectra of electrons, hydrogen, and helium nuclei over the corresponding energy intervals; studied the energy spectra, time variations and spatial gradients associated with recurrent and non-flare-associated interplanetary proton and helium streams; and determined the chemical compositions of both solar and galactic cosmic rays. The experiment performed normally during the lifetime of the spacecraft.

Cosmic Ray Experiment (University of Chicago–Dr. J.A. Simpson)

This experiment was designed to measure energy spectra, nuclear composition, and electrons over a wide range of energies and a wide dynamic range of fluxes. The energy range extended from ~ 0.5 MEV to > 1200 MEV/nucleon. The dynamic flux range was at least $\sim 10^5$. Emphasis was placed on high charge resolution over a wide dynamic charge range extending to $Z \sim 30$, and a high resolution for the light isotopes of hydrogen, helium and lithium. Emphasis was placed on the large dynamic ranges mentioned above and establishing a priority for rare particle events in the cosmic ray and solar particle fluxes. The composition telescope failed within a day after launch, but the performance of the other experiment systems was normal through the IMP I mission.

Low Energy Proton and Electron Differential Energy Analyzer (University of Iowa–Dr. L.A. Frank)

The objectives of this experiment were to measure the differential energy spectrums, angular distributions, spatial distributions, and temporal variations of electrons and protons over an energy range from 5 ev

to 50 kev over the geocentric radial distance range of 1.03 to 30 Earth radii. Experiment operation was normal during the lifetime of the spacecraft.

Energetic Particles (University of California-Dr. K.A. Anderson)

The Energetic Particles Experiment and four detectors were designed to study: (1) acceleration of electrons at the Sun and their ejection into planetary space; (2) the propagation of these electrons in the interplanetary medium; (3) electron fluxes in the Earth's distant radiation zone, and (4) to monitor solar electron events. The experiment performed normally until the spacecraft mission was completed.

Solar Proton (Applied Physics Laboratory (APL)-Dr. C. Bostrom)

The solar proton monitoring experiment consisted of five separate detectors, each using one or more solid-state detector elements. The primary objective of this experiment was to make continuous systematic measurements of solar proton intensity, and to disseminate this information rapidly to the scientific community. Performance of this experiment was normal during the lifetime of the spacecraft.

Solar and Distant Magnetosphere Electrons (Goddard Space Flight Center-Dr. T.L. Cline)

This experiment using a crystal scintillator studied electrons from the non-relativistic to the relativistic region in the 50 keV to 2 MEV kinetic energy interval. The primary objectives were to: determine cosmic ray characteristics of electrons and positrons in this energy interval; and to measure their intensity and the velocity spectrum and its changes with time, and the anisotropies of each beam of solar electrons observed over this energy interval. The shock, transition region, magnetospheric tail and boundary electrons in this energy interval and the interplanetary medium and its effects on cosmic rays were also studied. The experiment continued to operate normally until the IMP I reentered.

Plasma Experiments

Plasma Detector Experiment (Goddard Space Flight Center-Dr. K.W. Ogilvie)

The purpose of the Plasma Experiment was to measure the bulk, velocity, density, and parallel and perpendicular temperatures of the

hydrogen and helium ions in the solar plasma within a 200 EV to 8 keV energy range. Initial experiment performance was normal for the first month. After one month, difficulty was encountered and the operation was ended.

Los Alamos-Sandia Plasma (Los Alamos Scientific Laboratory of the University of California—Dr. S.J. Bame)

A hemispherical electrostatic analyzer was used to study electrons and positive ions in order to complete descriptions of the particle populations in the solar wind, magnetosheath, and magnetotail. Objectives also were to study the bow shock and magnetopause boundaries separating the three major plasmas and other important problems relating to solar wind; the average direction of flow; bow-shock-generated perturbations and accelerated particles; thermal anisotropies and their relationship to the magnetic field; field-particle discontinuities; and sector structure. The experiment continued to operate during the lifetime of the IMP I mission.

Electric and Magnetic Field Experiments

DC Electric Fields (Goddard Space Flight Center—Dr. T.L. Aggson)

This experiment was designed to monitor the electric field vector using three orthogonal planes of extendible antennas mounted on the spacecraft. A high resolution analog-to-digital converter and a spectrometer processed the induced signals to obtain direct current and low frequency alternating current electric field information. Data was collected during the entire IMP I mission.

AC Electric Field, (University of Iowa—Dr. D.A. Gurnett and University of Minnesota—Dr. P.S. Kellogg and Goddard Space Flight Center—T.L. Aggson)

The scientific objectives of the University of Iowa Experiment were to study the origin and characteristics of naturally occurring radio noises, and measure them over the frequency range from 20 Hz to 200 Hz. Polarization, direction of arrival, and both electrostatic and electromagnetic propagation modes of naturally occurring radio noises were also investigated.

The GSFC Electric and Magnetic Fields Experiment was designed to measure electric fields in the dc to 200 kHz range in the magnetosphere and interplanetary space. The GSFC group measured a low frequency

range below 20 Hz, and the Universities of Iowa and Minnesota measured the frequencies above 2 Hz. The performance of both experiments was normal during the IMP I mission.

Magnetic Field (Goddard Space Flight Center-Dr. N.F. Ness)

The Magnetic Field Experiment was designed to measure the vector magnetic field in space with high accuracy and precision. The detector was a boom-mounted triaxial fluxgate magnetometer with four ranges. The major areas of investigation were the interplanetary magnetic field; magnetic field in the magnetosheath and its boundaries; Earth's magnetic tail field; and Earth's magnetosphere. Experiment performance was normal over lifetime of the mission.

Radio Astronomy

Radio Astronomy Experiment (University of Maryland-Dr. W.C. Erickson, co-investigators, Dr. F.T. Haddock, University of Michigan and Dr. R.G. Stone, GSFC)

The Radio Astronomy Experiment consisted of two completely independent, but complementary, radiometer systems using two separate dipole antennas. The University of Maryland/GSFC impedance probe and radiometer consisted of two swept receivers each independently programmable and each possessing an internal noise source for calibration purposes. The objective of this experiment was to study the spectra of the galaxy, the Sun, and Jupiter with high flux resolution. A radiometer, operating in either a stepping mode (eight frequencies) or at a single frequency, was connected to a 91-m dipole antenna. It covered a frequency range of 30 kHz to 2 MHz, which was accomplished by stepping the swept receiver at approximately 36 kHz per step for 16 steps and approximately 55 kHz per step for another 16 steps. Experiment performance was normal.

The University of Michigan Radiometer had two modes of operation, a frequency stepping mode and a single frequency mode. The frequency stepping mode was controlled by internal logic triggered by a timing signal from the spacecraft. The change from this mode to the single frequency mode was controlled by a ground command. This experiment was designed to study the radio spectra of the galaxy, the Sun, and Jupiter with relatively high time resolution. It operated normally during the lifetime of the mission.

IMPs H AND J

Engineering Tests

Thermal Coatings (Goddard Space Flight Center-P. Maag)

This test was designed to measure several thermal coatings for changes in solar thermal energy absorbency due to exposure to the space environment. Twelve samples of thermal coatings, one of which was a black reference, were mounted on the spacecraft body, but thermally isolated from it. The temperatures were monitored by thermistors and information telemetered to the ground during the operation of the IMP H spacecraft.

Data System Test (DST) (Goddard Space Flight Center-T. Goldsmith)

The DST consisted of a data multiplex unit (DMU), a data processing unit (DPU), and a power converter. The objective of the engineering test was to flight-qualify the DMU and the DPU as central data handling equipment on future spacecraft. The DST was used by several IMP J experimenters to process on board the experiment's raw data, and transmit the pulse-code-modulated data via the analog transmitter.

The DMU in IMP J differed from that in IMP H most significantly in that new ion-implanted p-channel metal oxide silicon microcircuits were used, and a new non-systematic 24-bit convolutional code system was used. The DPU was similar to that used in the IMP H spacecraft.

Solar Cells (Goddard Space Flight Center-N. Mejia)

Solar cell engineering tests were made on IMP H to determine the operating efficiency of integral glass solar cells. The cells were mounted on a solar panel similar to those of the solar cells used to obtain operating power. The integral solar cells used in this test were developed using new fabrication techniques which reduced the time of fabrication and manufacturing costs significantly with no apparent loss of efficiency. The current from the integral glass solar cells and their temperature characteristics were monitored and compared with those of the other solar cells in use on the spacecraft as part of the power supply system.

Solar Panel Test (Goddard Space Flight Center-E. Gaddy)

The purpose of this test on IMP J was to qualify a new cell for use in solar arrays for future missions. One of the 48 solar panels was fabri-

cated using new high-efficiency violet solar cells developed by Dr. Lindmayer, Communications Satellite (COMSAT) Corporation. The new cell had an efficiency of about 13.5 percent as compared with 11 percent for the commercial cells used on previously launched spacecraft. Tests made with the IMP J indicate the new cell, if used in future spacecraft solar arrays, would result in an increase in the power-to-weight ratio.

Scientific Experiments

Cosmic Ray Experiment (Goddard Space Flight Center—Dr. F.B. McDonald)

This experiment was a part of a systematic program to study solar and galactic electrons and nuclei throughout the solar cycle using the IMP H and J spacecraft. Data from this experiment were used to study solar modulation, quiet-time and flare-associated anisotropies, solar and magnetospheric particle acceleration processes, and solar composition. The experiment consisted of four separate telescopes made up of various combinations of dE/dx and E detectors, including scintillators, surface barrier, and lithium-drifted silicon detectors. Nuclei were identified over a wide range of energies and charges (protons to ion nuclei). Electrons were identified from 150 keV to 15 MEV and isotopes of hydrogen and helium were measured in the range of 4 to 80 MEV/nucleon.

Cosmic Ray (University of Chicago—Dr. J.A. Simpson)

This experiment was designed to study solar flare particle acceleration and particle containment in the vicinity of the solar wind. It also measured energy spectra, nuclear composition, and electrons over a wide range of energies and fluxes using a solid-state telescope on IMP H and J. The energy range for nucleons extended from ~ 0.5 MEV to ~ 1200 MEV/nucleon. The flux dynamic range was at least 105. The energy range for electrons was primarily 0.3 to 10 MeV. This experiment operated normally during the life of IMP H and is still operating on IMP J.

Ion and Electron (University of Maryland—Dr. G. Gloeckler)

The objective of this experiment was to determine the composition and energy spectra of low energy particles observed during solar flares and 27-day solar events. Special emphasis was placed on charge measurements, particularly those of positively charged particles in the energy band 100 KeV/nucleon to 10 MeV/nucleon which had not been

previously measured. The instrumentation determined signs and magnitudes of charges, measured energies of cosmic ray particles, and identified positrons, electrons, protons, and helium and certain other nuclei. An electrostatic analyzer separated the particles depending on their energy/charge ratios. The particles were detected and energy measured by a combination of solid state detectors. A thin window proportional counter/solid state detector dE/dx versus E telescope extended the energy range to 10 MeV/nucleon and permitted chemical and isotope composition measurements during the IMP mission. The IMP J instrument remains operational.

Solar Electrons (Goddard Space Flight Center-Dr. T.L. Cline)

This experiment was flown only on IMP H and it was designed to study solar flare X-rays and interplanetary electrons and positrons from the nonrelativistic to the relativistic regions. Energy spectra, intensities, and variations with solar and magnetic activity were studied to determine whether they have cosmic ray characteristics. Other objectives were to study shock phenomena, transition regions, and magnetospheric tail and boundary electrons. Two collimated electron detectors (a scintillator and an anti-correlation scintillator) mounted at right angles, with a single background detector mounted nearby, comprised the basic instrumentation. The instrument measured solar flare X-rays from 20 keV to 1 MEV and electrons and positrons from 100 keV to 2 MEV during the lifetime of the IMP H spacecraft.

Electrons and Isotopes (California Institute of Technology-Dr. E.C. Stone)

The purpose of this experiment was to obtain data to study local acceleration of particles, solar particle acceleration processes, and particle storage in the interplanetary medium and to study the interstellar propagation and solar modulation of particles in the interplanetary medium. This was accomplished by measuring the differential energy spectra of electrons and hydrogen and helium isotopes.

The instrumentation consisted of a multi-element, totally depleted solid-state telescope with anti-coincidence shielding. The detector was specially designed to eliminate scattered electron effects.

Energetic Particles (National Oceanic and Atmospheric Administration-Dr. D.J. Williams)

This experiment was designed to study the propagation characteristics of solar cosmic rays through the interplanetary medium; to study

electron and proton patches throughout the geomagnetic tail and near and through the flanks of the magnetopause; and to study the entry of solar cosmic rays into the geomagnetic field. This was accomplished by comparing data from measurements obtained with IMP H and J to similar data from the TIROS series of satellites.

The instrumentation consisted of a three-element telescope configuration employing solid-state detectors and a magnetic field to deflect electrons. Two side-mounted detectors were used to detect the electrons deflected by the magnet.

Charged Particles (John Hopkins University Applied Physics Laboratory—Dr. S.M. Krimigis)

The main objective of this experiment was to measure concentrations of protons, alpha particles, heavier nuclei, and X-rays in a wide energy interval. The experiment was primarily designed to study solar radiations, but with sufficient dynamic range and resolution to measure cosmic rays and magnetospheric tail particles. The data obtained was used to study angular distributions, energy spectra, propagation of particles emitted from the Sun, as well as those streaming along the magnetospheric tail away from the Earth.

A set of three solid-state detectors placed inside a scintillator cup was used. Thin-window Geiger-Müller tubes inside an anti-coincidence scintillator were also used to obtain measurements of low-energy electrons and solar and galactic X-rays. The experiment was successful on both IMP H and J missions, and remains operational on IMP J.

Plasma (Massachusetts Institute of Technology—Dr. H.S. Bridge)

This experiment was intended to measure plasma properties in the interplanetary region, the transition region, and in the tail of the magnetosphere during both IMP H and J missions. Parameters measured included energy distributions and also angular distributions of electrons in the equatorial plane of the spacecraft. The flow direction relative to the spacecraft equatorial plane was also determined.

The basic instrumentation consisted of a split collector Faraday Cup (split about the equatorial plane of the spacecraft) with a modulation potential applied to one of the grids.

Low Energy Particles (University of Iowa—Dr. L.A. Frank)

This experiment was designed to study the differential energy spectra of low energy electrons and protons measured over the geocentric radial

distance of 40 earth radii, to increase the understanding of geomagnetic storms, aurora, tail and neutral shield, and other magnetospheric phenomena.

The basic instrument was a dual-channel, curved-plate electrostatic analyzer employing continuous channel multiplier systems in conjunction with an Anton-type 213 Geiger-Müller tube. The instrument measured electrons and protons over the energy range of 5 EV to 50 EV during the IMP H and continues operational on the IMP J mission.

Ion Composition (Goddard Space Flight Center-Dr. K.W. Ogilvie)

The object of this experiment was to measure the ion composition of the solar wind and relate the information to the temperature and composition of the solar corona and photosphere. The experiment, flown only on IMP H, investigated plasma energization processes, mass-to-charge ratios, and energies of solar wind ions with velocities between 200 and 600 kilometers per second. Heavy ions of helium and oxygen were identified. An electro-static analyzer was used in conjunction with a cross-field velocity selector and a spiraltron-counting detector.

Plasma (Los Alamos Scientific Laboratory-Dr. S.L. Bame)

This other experiment was to obtain data to make a comprehensive study of electrons and positive ions in the IMP H orbital path, and to coordinate this data with the magnetometer and other data obtained. The instrumentation was capable of detecting ions at least as heavy as oxygen, and also separately identifying them in the energy-per-charge spectrum when solar wind ion temperatures were low.

The basic instrument was a plasma analyzer which consisted of a set of hemispherical analyzer plates and an electron multiplier together with associated detector electronics, voltage supplies, and logic circuits. The analyzer measured electrons in the energy range of 4 to 20,000 EV and protons in the energy range of 70 to 20,000 EV.

Magnetic Fields (Goddard Space Flight Center-Dr. Norman F. Ness)

This experiment, flown on IMP H and J, measured vector magnetic field in three dynamic ranges: ± 12 , ± 36 and ± 108 gammas. The three mutually orthogonal fluxgate sensors were located at the ends of a 3-meter instrument boom. Data from the three sensors were used to study the interplanetary magnetic field, the Earth's magnetic tail, and the interaction of the solar wind with the geomagnetic field.

Plasma Wave (TRW, Inc.-Dr. F.L. Scarf)

The purpose of this IMP H experiment was to measure two components of the vector electric field and one component of the vector magnetic field in eight discrete frequency intervals in the range of 10 Hz to 100 Hz. Instrumentation consisted of a 61-cm electric dipole antenna and an 18-cm magnetic loop antenna mounted on a 3-meter instrument boom. Measurements were made of the electric and magnetic field components of plasma waves in the solar wind, in the bow shock and transition regions and in the geomagnetic tail. Results were correlated with particle data taken from other spacecraft instruments.

DC Electric Fields (Goddard Space Flight Center-Dr. T.L. Aggson)

This experiment was flown on IMP J to measure DC electric field vectors with a sensitivity of about 0.1 mv/meter using a biaxial antenna system. Electrometers measured the potential difference between the two halves of each antenna, permitting exploratory electrostatic field measurements in the solar wind, in the transition region, and in the geomagnetic tail. The results of the DC electric field experiment were correlated with plasma and magnetic field measurements made on the spacecraft. The experiment was designed to operate at frequencies ranging from DC to 100 Hz and designed to investigate electric fields on the order of 10 uv/meter for AC signals and 100 uv/meter for DC signals.

AC Electric and Magnetic Fields (University of Iowa, Dr. D.A. Gurnett)

An array of electric and magnetic field anatennas, associated receivers and bandpass filters were flown on IMP J to measure spatial and temporal characteristics of both electric and magnetic AC vector fields and their polar relationships along the IMP J orbit. Electric fields were measured in the range of 0.3 to 59 Hz and electric and magnetic fields were measured in the range of 20 Hz to 200 kHz. The purpose of the experiment was to develop a better understanding of plasma dynamics, the Earth's shock front, and acceleration of particles.

Chapter 5

Engineering Accomplishments and Scientific Achievements

The series of IMPs launched from November 1963 to October 1973 provided the first accurate measurements of the interplanetary magnetic field, the magnetosphere boundary, and the shock wave associated with the interaction of the geomagnetic field and solar wind. Detection of the extended geomagnetic tail (plasma sheet) represented the most important result with respect to the Earth's magnetic field.

The two Anchored Interplanetary Monitoring Platform spacecraft, AIMPD (Explorer 33) and AIMPE (Explorer 35), launched in 1966 and 1967, respectively, made significant contributions to scientific knowledge and understanding of the near-lunar and interplanetary environment. One of the major findings was that the moon has no large-scale magnetic field and that a solar wind void exists behind the moon.

The successful IMP missions were the result of many scientific and engineering achievements. The following paragraphs detail some of these accomplishments for each IMP satellite in the series.

IMPs A, B, AND C

IMP A provided the first highly accurate direct measurements of interplanetary magnetic fields, and the magnetic field experiment carried on IMP A produced data for the first detailed mapping of the Earth's magnetic field on the night-time side of the magnetosphere.

Another important early finding from IMP A spacecraft data was the discovery of a detached bow shock wave preceding the Earth's magnetosphere. This was the first experimental evidence for the existence of a collision-less magneto-hydrodynamic shock wave. Figure 50 diagrams the Magnetic Field Experiment results. An artist's concept of the shock wave is shown in Figure 51.

The IMP A spacecraft also discovered a neutral sheet (a region with a very weak or non-existent magnetic field) which appears to serve as a repository for energetic particles.

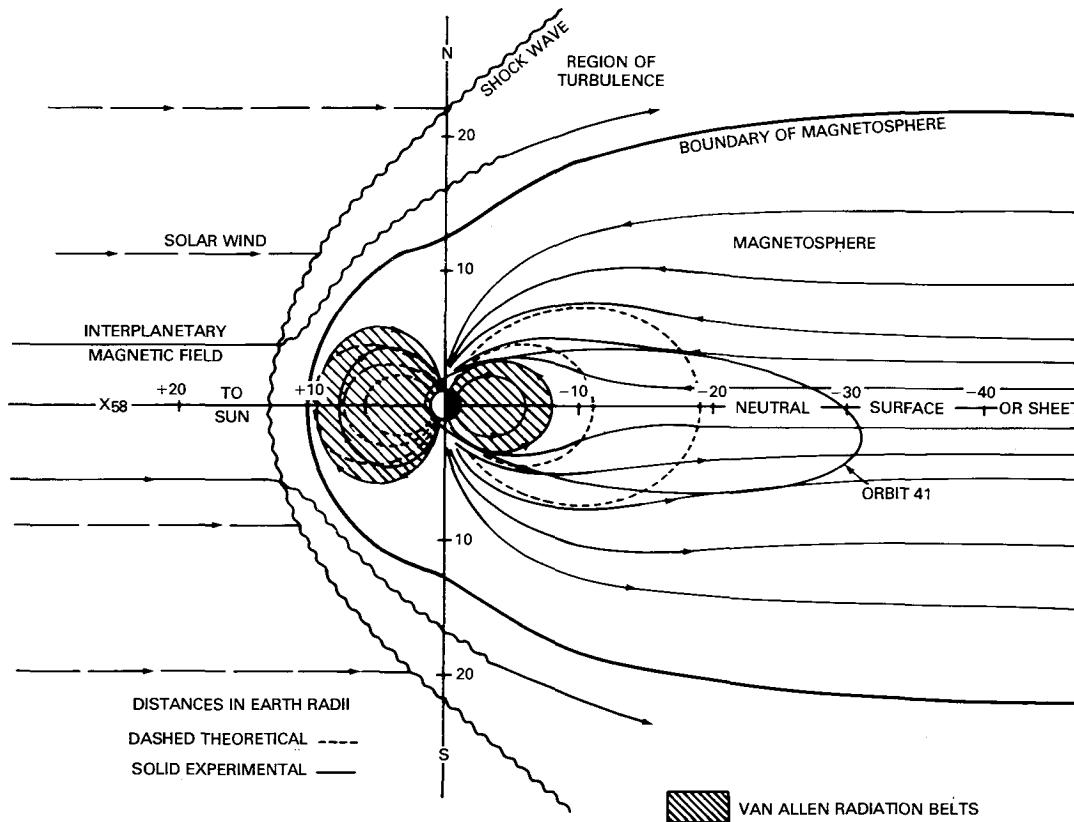


Figure 50. A Diagram of the Collisionless Magneto-Hydrodynamic Shock Wave Obtained from IMP A Magnetic Field Experiment (Noon-Midnight Meridian Plane Projection).

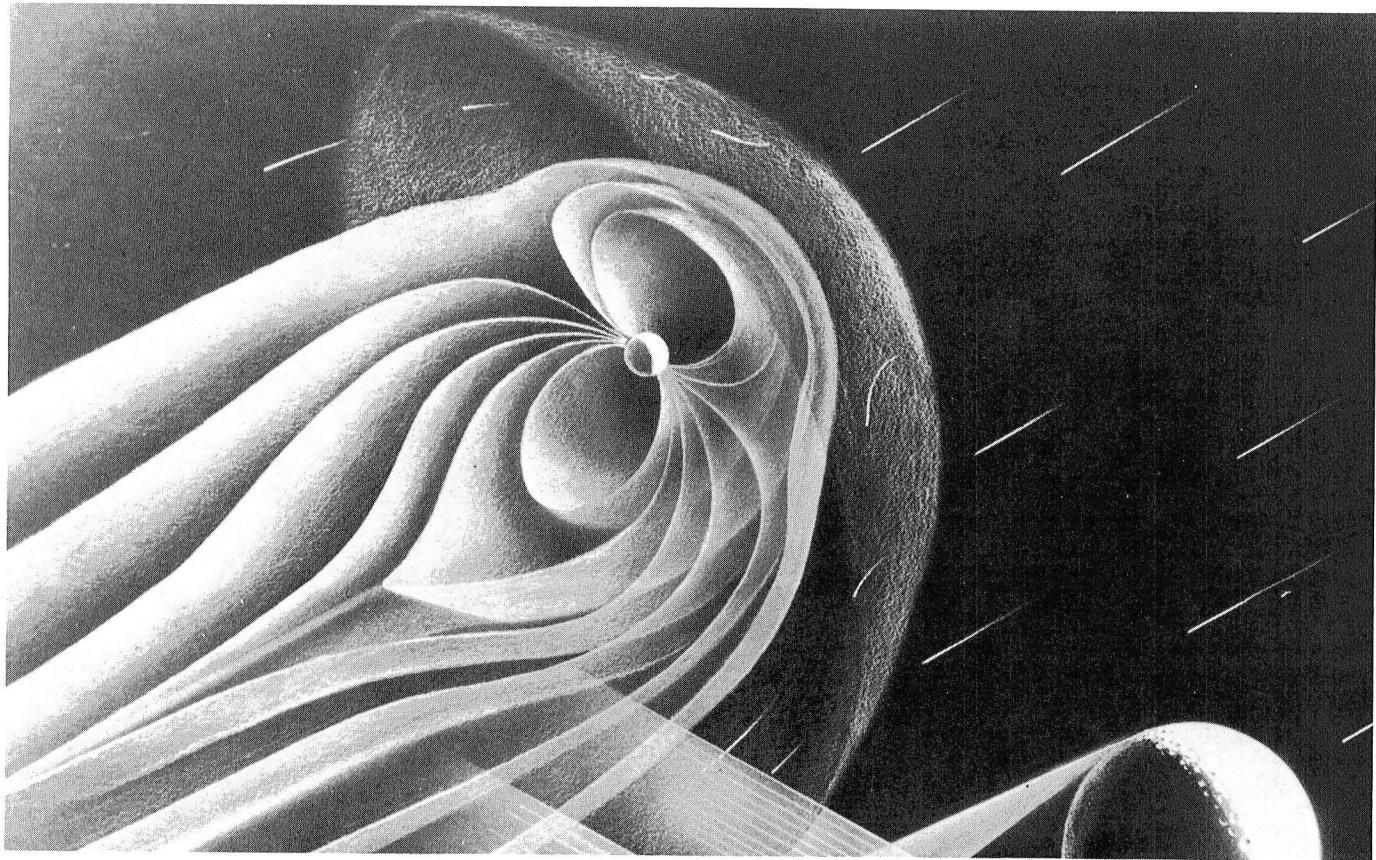


Figure 51. Artist's Concept of Collisionless Magneto-Hydrodynamic Shock Wave

IMP A provided the first extended opportunity to "see" far beyond the main region of the Van Allen radiation belts, lying about 64,370 km from the Earth, and map both near and distant radiation patterns around the entire Earth. Over 6000 hours of scientific data were transmitted over an 18-month period.

Another significant result of the IMP A measurements was the discovery of an extended "magnetic tail" behind the Earth caused by interaction of the solar wind with the geomagnetic field.

IMP B continued the mapping of the Earth's near-space environment and it was discovered that the solar wind interaction with the geomagnetic field was not a controlling factor for replacing particles in the magnetosphere.

IMP C, because of its highly eccentric orbit, provided additional data on the Earth's magnetosphere and "magnetic tail".

An optical aspect computer card composed of an integrated circuit was designed and developed by GSFC personnel and was used in IMPs A, B and C. This card was the first NASA integrated circuit to be put into orbit on a satellite.

Solid-state integrated circuits developed for the IMP series of spacecraft provided many advantages, including small size and weight and the need for fewer external connections. For example, using conventional technology the assembly of a two-transistor binary counter on a conventional printed circuit board required at least 22 connections. However, with an integrated circuit, the same function was performed with only six connections. This technology also allowed one complete system to be mounted on a single circuit board whereas in earlier spacecraft systems, three cards were required to perform the same function using conventional transistor circuitry.

One difficulty with measuring magnetic fields was the contamination of the measurements caused by fields originating on the space platform itself. Extensive efforts by electrical and mechanical engineers, the GSFC technical staff and those with experiments on-board the IMP A satellite reduced the magnetic contamination problems. As a result, IMP A was determined to be the magnetically cleanest of any United States satellite launched.

Prior to the IMP program, satellites, their experiments, and associated electronic packages were independently configured, thus restricting the number of experiments which could be mounted on the structural platform. In the IMP series, a new concept of experiment and electronic packaging called "modular packaging" was developed and used for the first time on IMP A. In this concept, all experiments and their associated

electronic packages are of the same shape but of varying height. The packages were placed into a structural frame that was similar to that of a chest of drawers. This allowed the packages to be easily removed from the system for repairs or replacement, even when the spacecraft was mated to the launch vehicle just before launch.

This same packaging concept also allowed a high density of experiments in a limited volume, with weights distributed to enhance spin stability with a minimum of balance weights. This concept was a major achievement in maximizing the useful payload weight, and was incorporated in all spacecraft in the IMP series.

Another significant engineering achievement resulting from the IMP program was the design of the wiring system used to connect spacecraft experiments, instrumentation, and associated electronics. In previous spacecraft, the wiring was accomplished with flexible interconnections to accommodate experiments that were of varying sizes and shapes and located at diverse locations on the structural platform. These interconnecting wiring cables were subjected to repeated flexing during the spacecraft integration process. This flexing caused many problems that required frequent repair of the wiring system.

With the concept of plug-in modules that were stacked in definite locations as drawers, fixed connectors were mounted rigidly on a rear wall directly behind each experiment. These fixed connectors allowed the removal of an experiment without disturbing the wiring. Consequently, the electronic integration personnel could fabricate a harness containing all the connecting cables (wires) into one bundle (harness) that could be fixed to the rear wall of instrument stacks and to the structural platform. The wiring was no longer flexed during testing phases, and many problems encountered with previous spacecraft were eliminated. This step was a significant improvement in the state-of-the-art of electronics integration.

AIMPs D AND E

As indicated earlier, the Anchored IMPs D and E were designed to be anchored to a lunar orbit. AIMP D did not achieve the lunar orbit intended although the later launched AIMP E did achieve the desired orbit.

For the AIMP D and E spacecraft, a nonmagnetic, electromechanically actuated oscillating mechanism called a triaxial fluxgate flipper was developed for the fluxgate magnetometer sensor assembly. This device provided the experimenters with an improved capability for remotely manipulating components. This compact, lightweight mechanism pro-

vided repeated bi-directional, rectilinear, and rotary forces over extended periods of time. One of these devices aboard the AIMPs D operated satisfactorily after having been actuated once a day for over 1,398 consecutive days. The induced magnetic field was less than one gamma at 5.08 cm from the sensors, and the residual magnetic field was less than 0.25 gamma at 7.62 cm.

In order to eliminate as many interface problems as possible, and to provide as much support as practicable to the experiment, a new encoder system was designed by GSFC engineers for AIMPs D and E. The encoder consisted of a number of cordwood-constructed modules with a new type of transistor called Metal Oxide Silicon Field Effect Transistor (MOSFET) chips used in the encoder. This repackaging produced a great savings in space and reduced the design complexity. It allowed AIMPs D and E to use encoders with a total of 4,500 semi-conductor devices packaged in less than 700 device packages whereas IMPs A, B and C launched earlier carried encoders with 1,200 transistors in 1,200 transistors packages (i.e., one transistor per package).

The use of MOSFET chips in the modular design of the AIMPs D and E encoders was a major step in the advancement of the state-of-the art in spacecraft electronics design. These devices were used in the follow-on spacecraft in the IMP series.

IMPs F AND G

A major engineering achievement for the IMP's F and G spacecraft was the electronic design and packaging of an advanced encoder and digital data processor (DDP) subsystem. This subsystem was designed and packaged so that both the encoder and the DDP were included in one package, thus saving both weight and valuable space. In addition, the experiment capacity of IMPs F and G was increased.

The DDP for IMPs F and G had a capacity of approximately 460 information bits. This capacity was four times that of the IMP A satellite. Also the digital bit rate was increased by a factor of ten over that employed by IMP A while maintaining the same transmitter power at the same range. This was accomplished by increasing the number of bits per channel from three to eight and increasing the channel rate by four.

In addition, design improvements for the IMP F spacecraft extended the useful life of that satellite well beyond the one-year design lifetime and permitted a delay in the launch of IMP G from March 1968 to June 1969. Since the IMP program consisted of a series of spacecraft launched on the average of one each year, the deferment of the IMP G

launch was equivalent to launching one less spacecraft in the 1968-1971 time period without compromising the program objectives.

IMP I

The IMP I spacecraft was the largest and most complex spacecraft to be completely built and integrated at the Goddard Space Flight Center. IMP I was the first IMP to incorporate body-mounted solar arrays and since it was launched in an orbit perpendicular to the ecliptic, a constant energy output was produced by these arrays. In addition, IMP I was the first IMP to measure spacecraft current by amplifying the voltage drop across a small (5 Milliohm) resistor in the power system. The practice of placing resistors in the power system circuits of earlier spacecraft had previously caused problems, and a bulky set of inductors, magnetic amplifiers, choppers, and rectifiers were needed to measure and telemeter this current.

Some of the other unique engineering features of the spacecraft included:

- The use of an encoder-digital data processor, which was the most advanced and powerful device of its type ever flown on an unmanned NASA spacecraft. It contained 328,000 metal oxide silicon field effect transistors (MOSFET) devices and 500 data counters, occupied 164 cubic centimeters and consumed only 4 watts of power;
- Incorporation of an on-board computer with a general-purpose, stored program containing features to permit time-sharing operations. This computer was used to process data from some of the scientific instruments on-board, and was the first of its kind to demonstrate the potential of the computer for use on future unmanned spacecraft missions and the capability for reprogramming from ground control.

IMPs H AND J

IMPs H and J constituted the climax in the series of ten highly successful IMP spacecraft. In operation as late as the Fall of 1979, IMPs H and J represented the state-of-the-art in spacecraft design, development, and engineering sophistication, when launched.

IMPs H and J contained a number of sensitive electromagnetic field experiments, as well as noise-producing experiments and digital systems. Because of this, isolation of various elements from interfering effects was exceedingly important. To meet this problem, GSFC engineers designed a new and unique wiring harness for IMPs H and J to

suppress conducted and radiated electrical and radio interference. The following techniques were used:

- Exceptionally sensitive or noisy leads were individually shielded.
- Other sensitive leads were routed in their own shielded bundle or subharness to minimize coupling.
- Special care in shielding and filtering was given to all harness components on the exterior of the main spacecraft structure.
- Cables from the experiment boom units plugged directly into the outside faces of the experiment spacecraft assembly to further minimize coupling.
- Isolation between power currents and signal leads was maintained by using power converters with isolated primary and secondary windings to avoid generation of stray magnetic fields by ground loops.

Because of these design features, wiring harnesses in IMPs H and J represented an advancement in the state-of-the-art for spacecraft wiring harnesses and established new techniques for use on future spacecraft.

Appendix A:

Spacecraft Summary

Descriptions

INTERPLANETARY MONITORING PLATFORM-A, EXPLORER 18

PROJECT MANAGER:	Mr. Paul Butler, Goddard Space Flight Center (GSFC).
PROJECT SCIENTIST:	Dr. Frank B. McDonald, GSFC.
LAUNCH:	26 November 1963 at 2130 hrs (EST) Eastern Test Range on Delta rocket.
APOGEE:	197,000 km.
PERIGEE:	192 km.
INCLINATION:	33.3 degrees.
PERIOD:	3 days, 22 1/2 hours.
VELOCITY:	38,800 km/hr at perigee. 1,250 km/hr at apogee.
WEIGHT:	62.35 kg.
MAIN STRUCTURE:	Octagon, 71.1 cm across the flats; 20.3 cm high.
APPENDAGES:	Four solar paddles, each 66.0 cm long by 45.7 cm wide; four antennas, 40.6 cm long; rubidium-vapor magnetometer, on 1.83 m boom; two fluxgate magnetometers on 2.13 m booms.
POWER SYSTEM	
Power Supply:	11,520 solar cells, mounted on four solar-oriented arrays; one 15-volt, five-ampere-hour battery pack of 13 silver-cadmium cells.

Voltage: 12 to 19.6 vdc.

Power: 38 watts.

COMMUNICATIONS AND DATA HANDLING SYSTEM

Telemetry: Pulse-frequency modulation (PFM).

Transmitter: 4-watt output at 136.110 MHz.

Encoder: PFM with digital data processor (DDP) for accumulation and storage of data.

TRACKING

Tracking Stations: Apogee—Johannesburg,
South Africa
Rosman, North Carolina
Carnarvon area of
Australia
Perigee—Blossom Point, Maryland
Fort Myers, Florida
Goldstone, California

DATA

ACQUISITION STATIONS: Johannesburg, South Africa
Wommera, Australia
Santiago, Chile

RANGE AND

RANGE RATE STATIONS: Scottsdale, Arizona

EXPERIMENTS

Magnetic Field Experiment: Rubidium-vapor magnetometer.
Dr. Norman F. Ness, GSFC

Magnetic Field Experiment: Fluxgate magnetometers.
Dr. Norman F. Ness, GSFC

Cosmic Ray Experiment: Range versus energy loss.
Dr. J.A. Simpson, Enrico Fermi
Institute, University of Chicago

Cosmic Ray Experiment: Energy versus energy loss.
Dr. Frank B. McDonald and
Dr. George Ludwig, GSFC

Cosmic Ray Experiment:	Ion chamber and Geiger counter tubes. Dr. Kinsey A. Anderson, University of California
Cosmic Ray Experiment:	Orthogonal Geiger Counter Telescope Array. Dr. Frank B. McDonald, GSFC
Solar Wind Experiment:	Low energy proton analyzer. Dr. John Wolfe, Ames Research Center
Solar Wind Experiment:	Plasma probe. Dr. Herbert S. Bridge, Massachusetts Institute of Technology
Solar Wind Experiment:	Thermal ion electron sensor. Robert Bourdeau and Gideon P. Serbu, GSFC

**INTERPLANETARY MONITORING PLATFORM—B,
EXPLORER 21**

PROJECT MANAGER:	Paul Butler, Goddard Space Flight Center.
PROJECT SCIENTIST:	Dr. Frank B. McDonald, Goddard Space Flight Center.
LAUNCH:	October 3, 1964, 2245:00.4 EST Eastern Test Range on Delta Rocket.
APOGEE:	95,400 km.
PERIGEE:	193 km.
INCLINATION:	33.5 degrees.
PERIOD:	1 day, 10 hours and 57 minutes.
WEIGHT:	61.42 kg.
MAIN STRUCTURE:	Octagon, 71.1 cm in diameter, 20.3 cm deep.
APPENDAGES:	Four solar paddles, 66 cm long by 45.7 cm wide.
	Four antennas, 40.6 cm long.

	Rubidium-Vapor magnetometer on 1.83 m boom.
POWER SYSTEM	Two Flux-Gate sensors on 2.13 m booms. Solar cells mounted on four solar-oriented arrays; 14-volts, 5-ampere-hour silver cadmium battery pack.
Power Supply:	
Voltage:	12 to 19.6 vdc.
Power:	38 watts.
COMMUNICATION AND DATA HANDLING SYSTEM	
Telemetry:	Pulse-frequency modulation (PFM).
Transmitter:	4-watts @ 136.145 MHz.
Encoder:	PFM with digital data processor (DDP) for accumulation and storage of data.
TRACKING	
Tracking stations:	Apogee— Johannesburg, South Africa Rosman, North Carolina Carnarvon area of Australia Perigee— Blossom Point, Maryland Fort Myers, Florida Goldstone, California
DATA ACQUISITION STATIONS:	Johannesburg, South Africa Woomera, Australia Santiago, Chile
RANGE AND RANGE RATE STATION:	Scottsdale, Arizona (Motorola) Rosman, North Carolina
THE EXPERIMENTS	
Magnetic Field Experiment:	Rubidium vapor magnetometer. Dr. Norman F. Ness, Goddard Space Flight Center

Magnetic Field Experiment:	Fluxgate magnetometers. Dr. Norman F. Ness, Goddard Space Flight Center
Cosmic Ray Experiment:	Range versus energy loss. Dr. J.A. Simpson, Enrico Fermi Institute, University of Chicago
Cosmic Ray Experiment:	Energy versus energy loss. Dr. Frank B. McDonald and Dr. George Ludwig Goddard Space Flight Center
Cosmic Ray Experiment:	Ion Chamber and Geiger counter tubes. Dr. Kinsey A. Anderson, University of California
Cosmic Ray Experiment:	Orthogonal Geiger-Counter. Telescope Array. Dr. Frank B. McDonald, Goddard Space Flight Center
Solar Wind Experiment:	Low energy proton analyzer. Dr. John Wolfe, Ames Research Center
Solar Wind Experiment:	Plasma probe. Dr. Herbert S. Bridge, Massachusetts Institute of Technology
Solar Wind Experiment:	Thermal ion electron sensor. Robert Bourdeau and Gideon P. Serbu, Goddard Space Flight Center

**INTERPLANETARY MONITORING PLATFORM-C,
EXPLORER 28**

PROJECT MANAGER:	Mr. Paul Butler, Goddard Space Flight Center (GSFC).
PROJECT SCIENTIST:	Dr. Frank B. McDonald, GSFC.
LAUNCH:	29 May 1965 at 0700 hrs (EST) Eastern Test Range on Delta Rocket.

APOGEE:	260,799 km.
PERIGEE:	205.2 km.
INCLINATION:	33.87 km.
PERIOD:	4 days, 3 hours, 19 minutes.
TOTAL WEIGHT:	58.258 kg.
MAIN STRUCTURE:	Same as IMP-A.
APPENDAGES:	Four solar paddles, 70.1 cm long by 45.7 cm wide. Remainder, same as IMP-A.
POWER SYSTEM:	Same as IMP-A.
COMMUNICATION AND DATA HANDLING SYSTEM	
Telemetry:	Same as IMP-A.
Transmitter:	4-watt output at 136.125 MHz.
Encoder:	Same as IMP-A.
TRACKING	
Tracking Stations:	Same as IMP-A.
DATA	
ACQUISITION STATIONS:	Same as IMP-A.
RANGE AND	
RANGE RATE STATIONS:	Same as IMP-A.
EXPERIMENTS:	Same as IMP-A.

INTERPLANETARY MONITORING PLATFORM AIM-P-D, EXPLORER 33

PROJECT MANAGER:	Mr. Paul Marcotte
PROJECT SCIENTIST:	Dr. Norman Ness
LAUNCH:	1 July 1966, Eastern Test Range, Florida on Delta 39 Vehicle.
APOGEE:	450,000 km.
PERIGEE:	50,000 km.
INCLINATION:	28.5°
PERIOD:	15.5 days.
TOTAL WEIGHT:	94.058 kg including 4th stage (retro-motor).

MAIN STRUCTURE:	Octagonal structure approximately 71.1 cm across the flats and 30.5 cm high, 87.9 cm in height from 3rd stage interface to top of 4th stage (retro-motor).
APPENDAGES:	Four solar arrays (paddles) each 70.1 cm long by 64.1 cm wide. Four antennas, tilted-turnstile, at 15° angle to top surface. Two fluxgate magnetometer sensors on opposing 2.13 m booms.
POWER SYSTEM:	7680 N-on-P solar cells on four solar-oriented paddle arrays. Average power supplied was 66.1 watts initially (49.0 watts at end of life) at 19.6 or 18.3 volts. Regulated power provided to 11 ampere-hour 15 volts silver-cadmium battery pack.

COMMUNICATIONS AND DATA HANDLING SYSTEM

Telemetry:	Pulse-frequency modulation (PFM).
Transmitter:	Minimum power output of 7 watts at a frequency of 136.020 MHz.
Receiver:	Phase modulation (PM) receiver with a sensitivity of -118 dbm for command and range and range rate reception.
Encoder:	PFM with a digital-data processor (DDP) for accumulation and storage of data.
On-Board Clock:	Unique crystal-controlled sequence counter.

Launch Vehicle:	Thrust-augmented improved Delta (DSV-3E) with an MB3-Block III Rocketdyne first-stage main engine, Aerojet-General AJ10-118-B liquid-propellant second-stage engine, a UTC-FW-4D third stage, and Thiokol TE-M-458 rocket motor 4th stage (spacecraft retrorocket).
Tracking Stations:	Until injection into orbit, Eastern Test Range and Space Tracking and Data Acquisition Network (STADAN); thereafter, GSFC STADAN.
Telemetry Receiving:	GSFC and STADAN.
EXPERIMENTS	
Engineering Tests:	Solar Cell Damage. L. Slifer, GSFC
Passive:	a. Lunar Ionosphere Effects on Radio Wave Propagation. A.M. Peterson, Stanford University, California
	b. Selenodetic Information. W.M. Kaula, University of California
Scientific:	Cosmic Ray Experiments a. Ion Chamber. K.A. Anderson, University of California
	b. Electrons and Protons Experiment. J.A. Van Allen, University of Iowa
	c. Low Energy Integral Spectrum Measurement Experiment. G.P. Serbu and E.J. Maier, GSFC

Magnetic Field and Plasma Experiments

- a. Ames Magnetometer
Experiment.
C.P. Sonett, Ames Research
Center, NASA
- b. Goddard Magnetometer.
N.F. Ness, GSFC
- c. Plasma Probe.
H.S. Bridge, Massachusetts
Institute of Technology

**INTERPLANETARY MONITORING PLATFORM (AIMP-E),
EXPLORER 35**

PROJECT MANAGER:	Mr. Paul Marcotte
PROJECT SCIENTIST:	Dr. Norman Ness
LAUNCH:	19 July 1967, Eastern Test Range, Florida on Delta 50 Vehicle.
APOSELENE:	9429 km.
PERISELENE:	2538 km.
INCLINATION:	169°.
PERIOD:	11.5 hours.
TOTAL WEIGHT:	104.32 kg including 4th stage (retro-motor).
MAIN STRUCTURE:	Same as AIMP-D.
APPENDAGES:	Same as AIMP-D.
POWER SYSTEM:	Same as AIMP-D.
COMMUNICATIONS AND DATA HANDLING SYSTEM	
Telemetry:	Same as AIMP-D.
Transmitter:	Minimum power output of 7 watts at a frequency of 136.110 MHz.
Receiver:	Same as AIMP-D.
Encoder:	Same as AIMP-D.
On-Board Clock:	Same as AIMP-D.
Launch Vehicle:	Same as AIMP-D.
Tracking Stations:	Same as AIMP-D.

Telemetry Receiving:	Same as AIMP-D.
EXPERIMENTS	
Engineering Tests:	Solar-Cell Damage, L. Slifer, GSFC. Attitude Control System, D. McCarthy, GSFC. Contamination Monitor, R. Sheehy, GSFC.
Passive:	Same as AIMP-D.
Scientific:	Cosmic Ray Experiments (same as AIMP-D). Magnetic Field and Plasma Experiments—same as AIMP-D plus: Micrometeorite Flux, J.L. Bohn and W.M. Alexander, Temple University

**INTERPLANETARY MONITORING PLATFORM IMP-F,
EXPLORER 34**

PROJECT MANAGER:	Mr. Paul Butler
PROJECT SCIENTIST:	Dr. Frank B. McDonald
LAUNCH:	24 May 1967, Western Test Range, California.
RE-ENTRY:	28 April 1969
APOGEE:	214,382 km.
PERIGEE:	242.1 km.
INCLINATION:	67.12 degrees.
PERIOD:	4 days, 7 hours, 51 minutes.
TOTAL WEIGHT:	73.92 kg.
MAIN STRUCTURE:	Octagonal structure when assembled with top cover, 71.1 cm across the flats, 28.6 cm in height.
APPENDAGES:	Two hinged fiberglass booms each 1.83 m in length; four solar cell paddles each 70.1 x 51.1 cm; four antennas, canted turnstile, at 25 degrees to top cover.

POWER SYSTEM:	6144, 2 x 2 cm, N-on-P solar cells on four solar-oriented paddle arrays; regulated power to non-magnetic silver-cadmium battery composed of 13 series-connected 5-ampere-hour Yardley cells.
COMMUNICATION AND DATA HANDLING SYSTEM	
Telemetry:	Pulse-frequency modulation (PFM).
Transmitter:	Minimum power output of 4 watts at a carrier frequency of 136.140 MHz.
Receiver:	Phase modulated (PM) receiver received commands at a frequency of 148.260 MHz for command and range and range rate reception.
Encoder:	Two-part system consisting of encoder and digital data processor (DDP) both housed in one package. DDP capacity (460 data bits) was four times that of IMPs A, B, and C.
On-Board Clock:	Unique crystal-controlled sequence counter.
Launch Vehicle:	Thrust-Augmented Improved Delta (TAID) Thor missile with three strap-on Thiokol solid-propellant motors, second stage AJ10-118-E liquid propellant, third stage FW-4 solid propellant rocket motor.

Tracking Stations:

Western Test Range (WTR) tracking and telemetry from lift-off to separation; Manned Space Flight Network (MSFN) Guaymas station provided second stage telemetry data acquisition and recording until loss of signal, then GSFC space Tracking and Data Acquisition Network (STADAN).

EXPERIMENTS

Engineering Test:

Radiation Damage Effects on Metal Oxide Silicon Field Effects Transistors (MOSFET), Mr. J.L. Wolfgang, Jr., GSFC.

Scientific:

Cosmic Ray and Energetic Particles Experiment

- a. Low Energy Telescope, Dr. W.L. Brown, Bell Telephone Laboratories.
- b. Ion Chamber, Dr. K. Anderson, University of California (Berkeley).
- c. Range Versus Energy Loss, Dr. J.A. Simpson, University of Chicago.
- d. Low Energy Proton and Electron Differential Energy Analyzer, Drs. J.A. Van Allen and L.A. Frank, University of Iowa.
- e. Cosmic Ray Anisotropy, Dr. K.G. McCracken, Southwest Center for Advanced Studies.
- f. Spherical Electrostatic Analyzer, Dr. F.B. Harrison, TRW Systems Group, TRW Inc.

- g. Solar Proton Monitoring Experiment, Dr. C. Bostrom Applied Physics Laboratory, and D.J. Williams and D.E. Guss, GSFC.
- h. Plasma Experiment, Drs. K.W. Ogilvie, GSFC, and T.D. Wilkerson, University of Maryland.
- i. Low Energy Proton and Alpha Detector, Dr. D.E. Hagge, GSFC.
- j. Energy Versus Energy Loss, Dr. F.B. McDonald, GSFC.

Magnetic Fields

- a. Magnetic Field Experiment, Dr. N.F. Ness, GSFC.

**INTERPLANETARY MONITORING PLATFORM IMP-G,
EXPLORER 41**

PROJECT MANAGER:	Mr. Paul Butler
PROJECT SCIENTIST:	Dr. F.B. McDonald
LAUNCH:	21 June 1969, Western Test Range, California.
RE-ENTRY:	23 December 1972
APOGEE:	176,826.8 km.
PERIGEE:	344.6 km.
INCLINATION:	86.8 degrees.
PERIOD:	3 days, 9 hours, 6 minutes.
TOTAL WEIGHT:	79.36 kg.
MAIN STRUCTURE:	Same as IMP-F.
APPENDAGES:	Two hinged fiberglass booms each 1.83 m in length. Differed from IMP-F in that one boom was a dummy appendage for purpose of balance only.
POWER SYSTEM:	Same as IMP-F.

**COMMUNICATION AND
DATA HANDLING SYSTEM**

Telemetry:	Same as IMP-F.
Transmitter:	Minimum power output of 4 watts at a carrier frequency of 136.080 MHz.
Receiver:	Same as IMP-F.
Encoder:	Same as IMP-F.
On-Board-Clock:	Same as IMP-F.
Launch Vehicle:	Same as IMP-F.
Tracking Stations:	Same as IMP-F.
EXPERIMENTS	
Engineering Test:	Same as IMP-F.
Scientific:	Cosmic Ray and Energetic Particles Experiments
	Same as for IMP-F, except the TRW Spherical Electrostatic Analyzer was removed and the following added:
a.	Low Energy Solar Flare Electron Detector, Dr. R. Lin, University of California (Berkeley).
b.	Low Energy Proton Differential Energy Analyzer (LEPDEA), Dr. L.A. Frank, University of Iowa, GSFC.

**INTERPLANETARY MONITORING PLATFORM IMP-I,
EXPLORER 43**

PROJECT MANAGER:	Mr. Paul Butler
PROJECT SCIENTIST:	Dr. F.B. McDonald
LAUNCH:	13 March 1971, Eastern Test Range, Florida.
RE-ENTRY:	October 1974

APOGEE:	206,258 km.
PERIGEE:	243 km.
INCLINATION:	28.69 degrees.
PERIOD:	4 days, 4 hours, 32 minutes.
WEIGHT (PAYLOAD)	288 kg.
MAIN STRUCTURE:	16-sided drum measuring 136 cm across flats and 182 cm in height.
APPENDAGES:	Two booms each approximately 1.5 m in length for attitude control system.
	Experiment booms approximately 8.5 m tip-to-tip.
	X-Y plane antennas—91 m tip-to-tip, Z (spin axis) plane antennas 3 m from spacecraft body.
POWER SYSTEM:	4,032, 2x6 cm, N-on-P silicon solar cells mounted on 48 panels on spacecraft outer surface.
	Regulated power to silver-cadmium battery containing 14 series-connected 10 ampere-hours Yardney cells.
COMMUNICATION AND DATA SYSTEM	
Telemetry:	Pulse-Coded-Modulation (PCM).
Transmitters:	<ul style="list-style-type: none">(1) One transmitter operated on a carrier frequency of 137.170 MHz, power output of 8 watts.(2) Another transmitter operated on a carrier frequency of 137.170 MHz with a power output of 4 watts.

Receivers:	<p>(1) PM signal for ranging encoder and AM signal for PCM command decoder. Operated on 148.260 MHz frequency.</p> <p>(2) Detected AM signal for sequential tone command decoder on 148.260 MHz aerospace telecommand frequency.</p>
Encoder:	<p>Split-phase readout PCM for transmitter modulation, used 200 logarithmic compressing accumulators, 120 binary accumulators and 600 bits of shift register storage. System was capable of storing 15,000 data bits.</p>
On-Board Clock:	Same as on IMP F.
Launch Vehicle:	Thrust-augmented McDonnell-Douglas modified Thor (DSV-2L-1B) booster, three strap-on Thiokol TX354-5 and three TX33-52 solid rocket motors; second stage (DSV-3E-4) was Aerojet-General Corporation AJ10-118-E with liquid propellants; third stage was spin-stabilized solid Thiokol Chemical Corporation rocket.
Tracking Stations:	Fairbanks, Alaska; Rosman, North Carolina; Fort Myers, Florida; St. Johns, Newfoundland; Winkfield, England; Johannesburg, South Africa; Tanarive, Madagascar; Quito, Ecuador; Santiago, Chile; Canberra, Australia; Carnarvon, Australia; and Kauai, Hawaii.

EXPERIMENTS

Engineering Test: On-Board Computer, Mr. R.A. Cliff, GSFC.

Scientific: Energetic Particles

- a. Cosmic Ray, Dr. F.B. McDonald, GSFC; co-investigators, Drs. B. Teegarden, GSFC, D. Hagge, Manned Spacecraft Center.
- b. Cosmic Ray, Dr. J.A. Simpson, University of Chicago (UC); co-investigators, Dr. M. Garcia-Monoz, S. Verma and Dr. J. Hsieh, UC.
- c. Low-Energy Proton and Electron Differential Energy Analyzer, Dr. L.A. Frank, University of Iowa.
- d. Energetic Particles, Dr. K.A. Anderson, University of California.
- e. Solar Protons, Dr. C. Bostrom, Applied Physics Laboratory (APL); co-investigators, Drs. D.J. Williams, GSFC and D.S. Beall, APL.
- f. Solar and Distant Magnetosphere Electrons, Dr. T.L. Cline, GSFC; co-investigator, Dr. K.A. Brunstein, Bendix Corporation.

Plasma

- a. Plasma, Dr. K.W. Ogilvie, GSFC.
- b. Plasma, Dr. S.J. Bame, Atomic Energy Commission, Los Alamos Scientific Laboratory, University of California (AEC-LASL-UC); co-investigator, Dr. J.R. Asbridge (AEC-LASL-UC).

Fields

- a. D.C. Electric Fields, Dr. T.L. Aggson, GSFC; co-investigator, Dr. J.P. Heppner, GSFC.
- b. A.C. Electric Fields, Dr. D.A. Gurnett, University of Iowa; co-investigators, Dr. P.J. Kellogg, University of Minnesota, Dr. T.L. Aggson and J.P. Heppner, GSFC.
- c. Magnetic Fields, Dr. N.F. Ness, GSFC; co-investigator, Mr. J. Seek, GSFC.

Radio Astronomy

- a. Radiometer and Impedance Probe, Dr. W.C. Erickson, University of Maryland, co-investigators, Dr. F.T. Haddock, University of Michigan and Dr. R.G. Stone, GSFC.

**INTERPLANETARY MONITORING PLATFORM IMP-H,
EXPLORER 47**

PROJECT MANAGER:	Mr. Paul Butler
PROJECT SCIENTIST:	Dr. Norman F. Ness
LAUNCH:	22 September 1972, Eastern Test Range, Florida.
STATUS:	Data acquisition stopped 30 September 1978; spacecraft is in operational standby mode.
APOGEE:	25,609 km.
PERIGEE:	235,609 km.
INCLINATION:	17.21 degrees.
PERIOD:	12 days, 8 hours.
WEIGHT (PAYLOAD):	390 kg.

MAIN STRUCTURE:	Drum-shaped, 16-sided, measuring 157 cm high and 136 cm across the flats; consists of aluminum honeycomb shelf supported by struts and a 46 cm diameter thrust tube 38 cm deep; experiment section enclosed by metallic cover and side panels.
APPENDAGES:	Two diametrically opposed experiment booms 3 m long and two attitude control system booms spaced 90 degrees from the experiment booms 1.2 m long; also, 8 RF antennas, four 54.6 cm, and four 47.0 cm long.
POWER SYSTEM:	Three rings of solar panels on the outer surface of the spacecraft; provides 130 watts of power at 28 volts during normal spacecraft operations.
COMMUNICATION AND DATA SYSTEM	
Telemetry:	Pulse-Coded Modulation (PCM) operated at 137.920 MHz, 2 watts.
Transmitters:	Two telemetry transmitters, multiplexed to a common antenna system; both transmitters operated at frequencies in the 136-137 MHz band. <ul style="list-style-type: none">(1) A 12-watt transmitter for PCM encoder data.(2) An 8-watt transmitter used primarily for ranging information was also used under emergency conditions for encoder data.

Receivers: (1) One receiver used to provide ranging signals for 8-watt transmitter; it detected PCM signals for ranging decoder and detected AM signals for PCM command decoder.

(2) One other receiver used to detect AM signals for the sequential tone decoder.

Encoder: Processed, stored and multiplexed data from each experiment until read out as split-phase PCM for transmitter modulation; used logarithmic compression accumulators, and register storage; encoder time-multiplexed a 400 or 1,600 bits per second data stream with synchronization words and convolutional parity coding; approximately 100 buffered synchronization pulses were supplied for spacecraft experiments.

On-Board Clock: Spin synchronous clock generated 2^n pulses per spacecraft rotation period dividing rotation period into $2^n - 1$ equal time intervals.

Launch Vehicle: Three stage Delta #90 vehicle.

(1) First stage was modified Thor liquid propellant booster incorporating six Castor strap-on solid fuel rocket motors.

(2) Second stage was liquid propellant pressure-fed engine, employing nitrogen tetroxide and aerozene 50 as propellants.

(3) Third stage was spin-stabilized solid-propellant motor. Spacecraft used a solid propellant kick motor to achieve desired orbit.

Tracking Stations: Spacecraft Tracking and Data Acquisition Network (STADAN) provided support for IMP-H mission; also, stations were used at Fairbanks, Alaska; Rosman, North Carolina; Fort Myers, Florida; Winkfield, England; Johannesburg, South Africa; Tananarive, Malagasy Republic; Quito, Ecuador; Santiago, Chile; Canberra, Australia; and Carnarvon, Australia.

EXPERIMENTS

Engineering Tests:

- a. Thermal Coatings, P. Maag, GSFC.
- b. Data Systems, T. Goldsmith, GSFC.
- c. Solar Cell, N. Mejia, GSFC.

Scientific

Energetic Particles

- a. Cosmic Ray, Dr. F.B. McDonald, GSFC.
- b. Cosmic Ray, Dr. J.A. Simpson, University of Chicago.
- c. Ion and Electron, Dr. G. Gloeckler, University of Maryland.
- d. Solar Electrons, Dr. T.L. Cline, GSFC.
- e. Electrons and Isotopes, Dr. E.C. Stone, California Institute of Technology.
- f. Energetic Particles, Dr. D.J. Williams, National Oceanic and Atmospheric Administration.

- g. Charged Particles, Dr. S.M. Krimigis, Johns Hopkins University, Applied Physics Laboratory.

Plasma

- a. Plasma, Dr. H.S. Bridge, Massachusetts Institute of Technology.
- b. Low Energy Particles, Dr. L.A. Frank, University of Iowa.
- c. Ion Composition, Dr. K.W. Ogilvie, GSFC.
- d. Plasma, Dr. S.J. Bame, Los Alamos Scientific Laboratory.
- e. Plasma Wave, Dr. F.L. Scarf, TRW Systems Group.

Fields

- a. Magnetic Fields, Dr. N.F. Ness, GSFC.

**INTERPLANETARY MONITORING PLATFORM IMP-J,
EXPLORER 50**

PROJECT MANAGER:

Mr. William R. Limberis

PROJECT SCIENTIST:

Dr. Norman F. Ness

LAUNCH:

25 October 1973, Eastern Test Range, Florida.

STATUS:

Operational with Data Acquisition continuing as of date of this book.

APOGEE:

288,672 km.

PERIGEE:

141,050 km.

INCLINATION:

28.64 degrees.

PERIOD:

11 days, 23 hours, 40.5 minutes.

WEIGHT(PAYLOAD):

401 kg.

MAIN STRUCTURE:

Essentially the same as the IMP-H spacecraft.

APPENDAGES:	Two experiment booms, each 3.4 m long. Four experiment antennas, each 61 m long. Two inertia booms, each 47 cm long. Other appendages as in IMP-H summary sheets.
POWER SYSTEM:	48 body-mounted solar panels each consisting of 68, 2x6 cm N-on-P silicon solar cells, initially provided approximately 150 watts of power at 28 volts during normal operation.
COMMUNICATION AND DATA SYSTEM	Silver-cadmium battery contained 14 series-connected 10-ampere-hour Yardney cells.
Telemetry:	Pulse-Coded Modulation(PCM).
Transmitters:	(1) Primary transmitter operated at frequency of 137.980 MHz, 12 watts. (2) Secondary transmitter operated at frequency of 136.800 MHz, 8 watts.
Receivers:	(1) One 148.980 MHz, for RARR interrogation, 64 PCM/FSK commands, and DST memory load commands. (2) One 148.980 MHz receiver for receiving standard GSFC tone commands.
Encoder:	Same as IMP-H.
On-Board Clock:	Same as IMP-H.
Launch Vehicle:	Same as IMP-H.
Tracking Stations:	Same as those used in tracking the IMP-H spacecraft.
EXPERIMENTS	
Engineering Tests:	<ol style="list-style-type: none"> a. Data System Test, T.C. Goldsmith, GSFC. b. Solar Panel Test, E. Gaddy, GSFC.

Scientific

Energetic Particles

- a. Cosmic Ray, Dr. F.B. McDonald, GSFC.
- b. Cosmic Ray, Dr. J.A. Simpson, University of Chicago.
- c. Energetic Particles, Dr. D.J. Williams, National Oceanic and Atmospheric Administration.
- d. Charged Particles, Dr. S.M. Krimigis, Johns Hopkins University, Applied Physics Laboratory.
- e. Electrons and Isotopes, Dr. E.C. Stone, California Institute of Technology.
- f. Ion and Electron, Dr. G. Gloeckler, University of Maryland.

Plasma

- a. Low Energy Particles, Dr. L.A. Frank, University of Iowa.
- b. Plasma, Dr. S.J. Bame, Los Alamos Scientific Laboratory.
- c. Plasma, Dr. H.S. Bridge, Massachusetts Institute of Technology.

Fields

- a. Magnetic Fields, Dr. N.F. Ness, GSFC.
- b. D.C. Electric Fields, Dr. T.L. Aggson, GSFC.
- c. A.C. Electric and Magnetic Fields, Dr. A. Gurnett, University of Iowa.

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Appendix C: Abbreviations and Acronyms

AC	Alternating Current
ACS	Attitude Control System
AIMP	Anchored Interplanetary Monitoring Platform
AM	Antenna Module; also, Amplitude Modulation
APP	Analog Performance Parameter(s)
BPP	Binary Performance Parameter(s)
BTL	Bell Telephone Laboratories
cm	centimeter(s)
CTA	Component Test Assembly
DC	Direct Current
DDP	Digital Data Processor(s)
D/P	Delta Instrumentation Package
DMU	Delta Multiplex Unit
DPU	Data Processing Unit
DST	Data System Test
EMI	Electromagnetic Interference
EFM	Electrostatic Field Measurement
EPS	Electrochemical Power Source
EST	Eastern Standard Time
ETR	Eastern Test Range
ETU	Engineering Test Unit
EV	Electron Volt
FM	Frequency Modulation
FSK	Frequency Shift Keying
FU	Flight Unit
GM	Geiger-Müller
GMT	Greenwich Mean Time
GRARR	Ground Range And Range Rate
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
GUM	Goddard/University of Maryland
hrs	hours

IMP	Interplanetary Monitoring Platform
k	kilo
keV	kiloelectron Volt(s)
kg	kilogram
km	kilometer
LED	Low Energy Detector
LEPEDEA	Low Energy Proton and Electron Differential Energy Analyzer
M	Mega; also million
m	meter
ma	milliampere
MEV	Million Electron Volts
MOSFET	Metal Oxide Semiconductor Field Effect Transistor
NASA	National Aeronautics and Space Administration
NSSDC	National Space Sciences Data Center
OA	Optical Aspect
PCM	Pulse Coded Modulation
PDP	Project Development Plan
PDT	Pacific Daylight Time
PF	Proto-Flight
PFM	Pulse Frequency Modulation
PM	Pulse Modulation
PP	Performance Parameter; also Project Plan
PPC	Performance Parameter Card
PU	Prototype Unit
RADEM	Radiation Damage Effects On MOSFET
RARR	Range and Range Rate
RF	Radio Frequency
RPM; rpm	Revolutions Per Minute
SCAS	Southwest Center for Advanced Studies
SSC	Spin Synchronous Click
STADAN	Space Tracking And Data Acquisition Network
SU	Space Unit
TAID	Thrust Augmented Improved Delta
UOI	University of Iowa
UT	Universal Time
V	Voltage
VLED	Very Low Energy Detector
WGT	Working Group Team
WTR	Western Test Range



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16. Abstract In the Fall of 1979, the last of ten Interplanetary Monitoring Platform Satellite (IMP) missions ended a ten-year series of flights dedicated to obtaining new knowledge of the radiation effects in outer space and of solar phenomena during a period of maximum solar flare activity.			
<p>The technological achievements and scientific accomplishments from the IMP program were numerous and many design "firsts" resulted. They began with the development and successful launch of the first IMP A satellite in November 1963 and ended with the IMP J satellite which censed its major data collection efforts in 1979. The series of IMPs provided the first accurate measurements of the interplanetary magnetic field, the magnetosphere boundary, and the shock wave associated with the interaction of the geomagnetic field and solar wind. Detection of the extended geomagnetic tail (plasma sheet) represented the most important result with respect to the Earth's magnetic field.</p> <p>The two Anchored Interplanetary Monitoring Platform spacecraft, AIM D (Explorer 33) and AIM E (Explorer 35), lauched in 1966 and 1967, respectively, made significant contributions to scientific knowledge and understanding of the near-lunar and interplanetary environment. One of the major findings was that the moon has no large-scale magnetic field and that a solar wind void exists behind the moon.</p>			
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